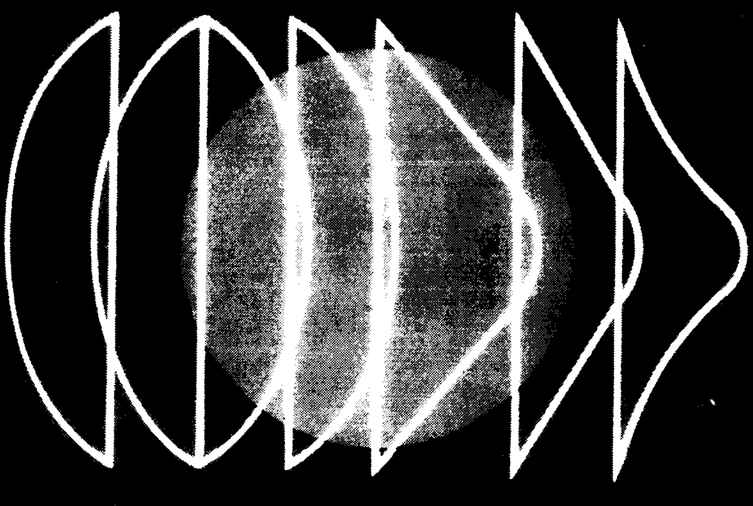
VOYAGER CAPSULE PHASE B FINAL REPORT



REPORT NO. F694
31 AUGUST 1967

VOYAGER CAPSULE PHASE B FINAL REPORT

VOLUME I SUMMARY

PREPARED FOR:
CALIFORNIA INSTITUTE OF TECHNOLOGY
JET PROPULSION LABORATORY
PASADENA, CALIFORNIA
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MCDONNELL ASTRONAUTICS

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SECTION 1

INTRODUCTION

The National Aeronautics and Space Administration and the Jet Propulsion Laboratory have projected to industry the challenge of the decade: to participate with them in making the first voyage to the surface of the planets. VOYAGER Capsule Phase B studies to " - select a single project approach from among the alternate approaches - " have been made and the results are reported in Volumes II through VI. This Volume I summarizes the key features of the resulting preferred Flight Capsule concept.

McDonnell Douglas Corporation and its subcontractors, General Electric Reentry Systems Division and Philco-Ford Space & Reentry Systems Division, make this report as part of JPL Contract 952000. We have utilized all applicable company resources in the execution of this preliminary design effort. Feasibility testing is essential to these studies so we have performed more than 100 separate test projects in our laboratories; see Figure 1-1 for a listing by test category.

The preferred systems described in this report have been designed to perform successfully the first time and every time. Guidelines and constraints designated by JPL have been met without exception. Performance requirements have been exceeded. Such constraints as accommodating the ten model atmospheres, landing on surface discontinuities of plus or minus 34 degrees slope, total 1973 Flight Capsule weight of 5,000 pounds containing a Surface Laboratory of at least 900 pounds, and the assumption that all extreme conditions will be encountered simultaneously have been among the most restrictive. These constraints have forced additional creativeness into the design so that in some instances, such as the lander configuration, it appears advisable to retain the resulting concept even if the constraints were found to be less demanding!

We have followed conservative design policies, using state-of-the-art components, functional redundancies with multiple paths to circumvent failures, contingency weight allocation to increase reliability, and stringent qualification requirements to increase confidence. The requirements for sterilization to meet planetary quarantine policies have been met. The reliability and economy of a standard design that remains essentially unchanged in subsequent flight opportunities have also been emphasized in making design decisions.

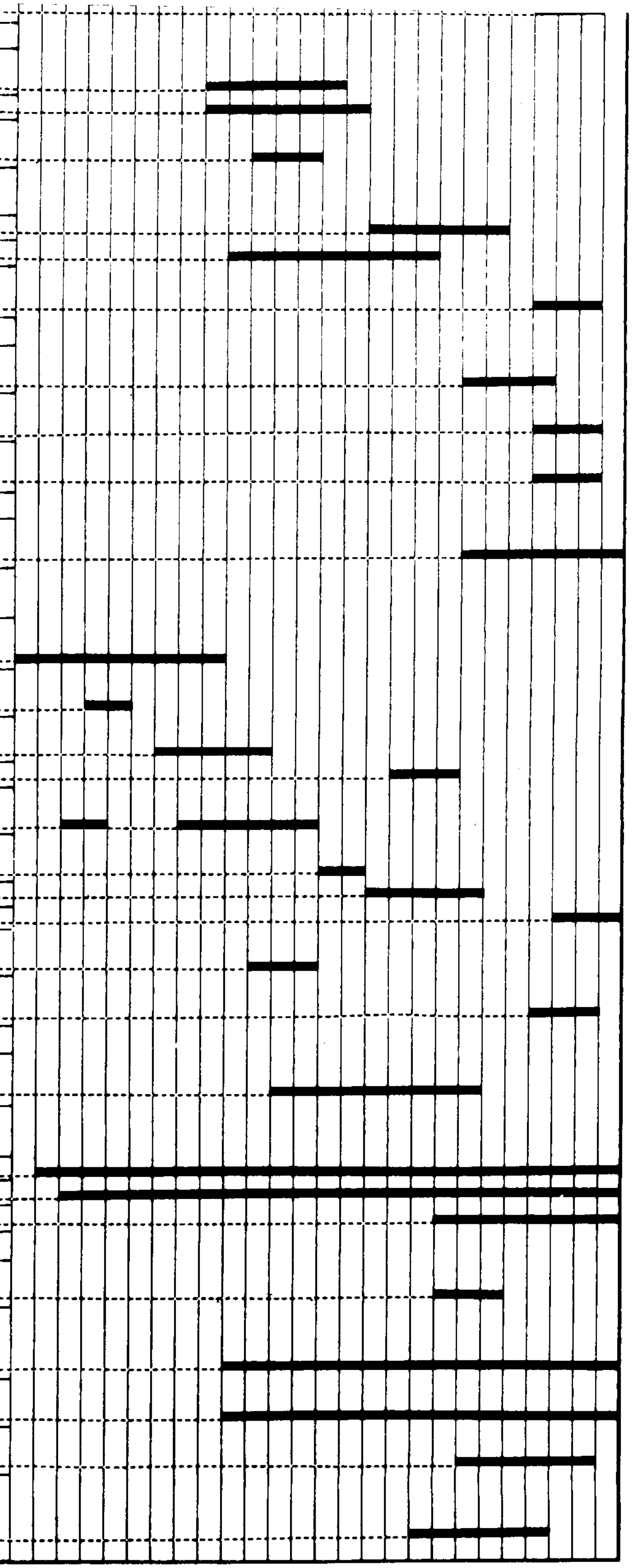
FEASIBILITY TEST PROGRAM

ESP	SLS	CBS	ACTIVITY
			MATERIALS EVALUATION
		X	Ablative Material Tests in Simulated Martian Entry Environments
		X	Effect of Sterilization and Vacuum Environments on Ablative Materials
		X	Ablative Material Weight Loss in a High Vacuum
	X	X	Effect of Time at Elevated Temperature on Energy Absorption Materials
	X	X	Evaluation of ORPEAM and Balsa-Cube Binder
	X	X	Lockalloy Mechanical Properties Test
	X	X	Effect of Sterilization on Resistance Welded Titanium
X	X	X	Long Term Material Exposure (ETO, Heat, Vacuum)
X	X	X	Accelerated Testing of Materials
		X	Heat Shield Materials Evaluation Program
			STRUCTURAL DESIGN EVALUATION
		X	Entry Cone Structural Panel Fabrication and Ablator Evaluation
		X	Static Test of Titanium Panels for Effect of Load on Ablator Band
		X	Static Test to Determine the Buckling Strength of Rings in 120° Conical Shapes
			LANDING SYSTEMS EVALUATION
	X	X	Sphere Penetration in Sand and Dust
		X	Erection Test of ¼ Scale Spherical Lander Model
		X	Deflection and Profile Tests on Flexible Inflated Torus Tank
		X	Drop Tests of ¼ Scale Torus Lander
		X	Overturning-Stability and Drop Tests of 1/10 Scale Legged Landers
		X	Determination of Stability of 1/10 Scale Uni-Disc Lander
		X	Development of Dynamic Impact Facility for Testing Energy Absorption Materials
		X	Static and Dynamic Energy Absorption Capabilities of Balsa Wood, Fiberglass Honeycomb, Aluminum Flexcore and Aluminum Crosscore Before and After Sterilization
			AERODYNAMICS
X		X	Entry Capsule Force and Moment Tests in Polysonic Wind Tunnel
X		X	Aerodynamic Decelerator Static Force and Moment Test in Trisonic Wind

Schedule: 1965 A S O N D; 1966 J F M A M J J A S O N D; 1967 J F M A M J J A S (months 1–26)

Figure 1-1

			Tunnel
			TELECOMMUNICATION
X	X	X	Antenna Breakdown Tests in Simulated Martian Atmospheres
X	X		S-Band Antenna Radiation Pattern Tests
X		X	Flow Field Seeding for Entry Ionization Suppression
X		X	Bit Synchronization in a Multipath Environment
X	X	X	RF Systems Multipaction Tests
X		X	RF Transmissibility Test
X	X	X	UHF Antenna Systems Radiation Pattern Tests
			PROPULSION
		X	Feasibility of Sterilizing Liquid Propellant Systems
		X	Material Compatibility During Sterilization of Anhydrous Hydrazine (N_2H_4)
		X	Evaluation of Sterilization Effects on Mono-Propellant Engine Performance
			PYROTECHNICS
	X	X	Effect of Sterilization and Long Storage Life on Pyrotechnic Actuation Devices
			SURFACE ENVIRONMENT SIMULATION AND THERMAL CONTROL EVALUATION
	X		Development of a Martian Environmental Simulation Facility
	X		Dust Particle Behavior in a Simulated Martian Atmosphere
	X		Behavior and Characteristics of Simulated Martian Sand and Dust Storms
	X		Wind Blown Sand and Dust Tests
	X		Effect of Voyager Mission Requirements on Thermal Control Coatings
	X		ETO Effects on Thermal Control Coatings
	X	X	Heat Pipe Demonstration
	X	X	Heat Pipe Control Valve Test
X	X	X	Measurement of Wind Velocity at Low Pressure
	X		Investigation of Martian Surface Phenomena
			RELIABILITY
X	X	X	Environmental Effects on Electronic Parts
			BIO-CONTAMINATION CONTROL TECHNIQUES EVALUATION
X	X	X	Microbiological Research
X	X	X	Sterile Assembly
X	X	X	Class 100 Facility Operation
			EXPERIMENTAL INVESTIGATION
X	X	X	Large Amplitude Coning in a Single Degree-of-Freedom Gyro
X	X		Efficiency of Molecular Separators for Interfacing a Gas Chromatograph With a Mass Spectrometer
X	X		Pyrolysis – Gas Chromatograph for Chemical Analysis
X	X		Digitization of Time-of-Flight Mass Spectra
X	X		Advantages of a Multi-Diameter Separation Column in Gas Chromatographic Analysis of Organics



The principal evaluation criteria that have been applied to the multiple candidates for each subsystem or mission mode and their weighting factors are:

Probability of Mission Success	0.35
System Performance	0.20
Development Risk	0.20
Versatility	0.15
Cost	0.10

These are discussed in Volumes II and III, Parts B, Sections 4 and 5.

Volumes II and III, which discuss the Capsule Bus and Surface Laboratory Systems respectively, have a similar format. The conclusions of our Phase B studies and the logic behind them are given in each volume in Part A, Preferred Design Concept and Part B, Alternatives, Analyses, Selection. Volume IV, Entry Science Package, discusses these same items in Parts D and E. Volume V, Interfaces, and Volume VI, Implementation, are subjects that involve relationships among all three systems. These subjects can be most clearly presented in separate volumes. This Volume I summarizes the other five.

The systems presented here fulfill the NASA/JPL requirements, meet the specified boundary conditions, and will perform the VOYAGER mission successfully.

SECTION 2

MISSION OBJECTIVES

The VOYAGER Program is a continuation and extension of the unmanned scientific exploration of the solar system. Its primary objective is to carry out scientific investigations by instrumented vehicles which will fly by, orbit, and/or land on the planets. The objectives of the missions to Mars, beginning in 1973, are to return information on the existence and nature of extraterrestrial life; on the atmosphere, surface, and body characteristics of the planet; and on the planetary environment. Experiments on the surface of Mars and in orbit about the planet will be performed to satisfy these objectives.

Specific goals for the 1973 mission are to:

a. Develop a system design capable of achieving a soft landing on the surface of Mars in 1973 and during the following opportunities.

b. Measure and transmit to Earth, via the Flight Spacecraft, atmospheric data and visual images of the planet during approach into the atmosphere and descent to the surface.

c. Obtain data on the Martian atmosphere and surface environment after landing and make initial measurements relevant to the question of the presence of life.

d. Develop a Surface Laboratory System design with communications and sequencing equipment that will be compatible with later VOYAGER missions.

e. Carry out surface operations for a period of at least one diurnal cycle (plus the time required to complete transmission of all acquired data).

Our Flight Capsule System design effort has been directed toward the system design of a standardized vehicle which will soft-land a variety of scientific payloads on the surface of Mars during the mission opportunities of 1973, 1975, 1977, and 1979. Since the later missions require a longer operating life on the Martian surface, the design and development phases of the 1973 study have emphasized compatibility with these requirements.

SECTION 3

DESIGN CRITERIA AND CONSTRAINTS

Operational Factors - The 1973 VOYAGER mission is constrained to operate within a launch window bounded by the Saturn V booster capability, the Spacecraft propulsion capability, and a minimum daily window, as shown in Figure 3-1.

The requirement to operate from orbits having periapse altitudes between 700-1500 km and apoapse altitudes between 10,000-20,000 km, plus the desire for landing site flexibility, establishes the design entry corridor shown in Figure 3-2. This corridor and the atmosphere definitions set the design conditions for the Capsule Bus subsystems. Other operational constraints imposed on the Flight Capsule design are tabulated in Figure 3-3.

Environmental Factors - The Mars atmosphere postulations which served as design boundaries for the Phase B study are shown in Figure 3-4; the surface environment is given in Figure 3-5.

Design Factors - Structural design and limit load factors are presented in Figure 3-6.

VOYAGER MISSION CONSTRAINTS

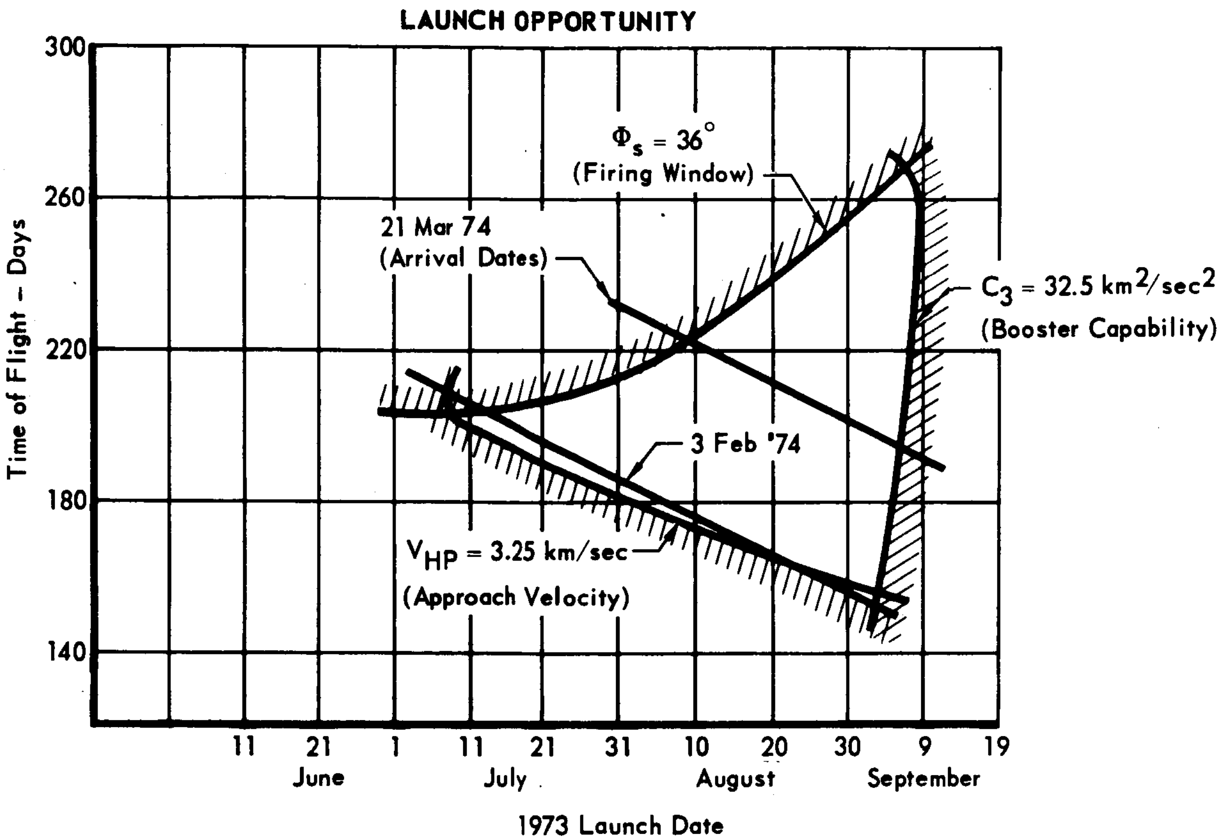


Figure 3–1

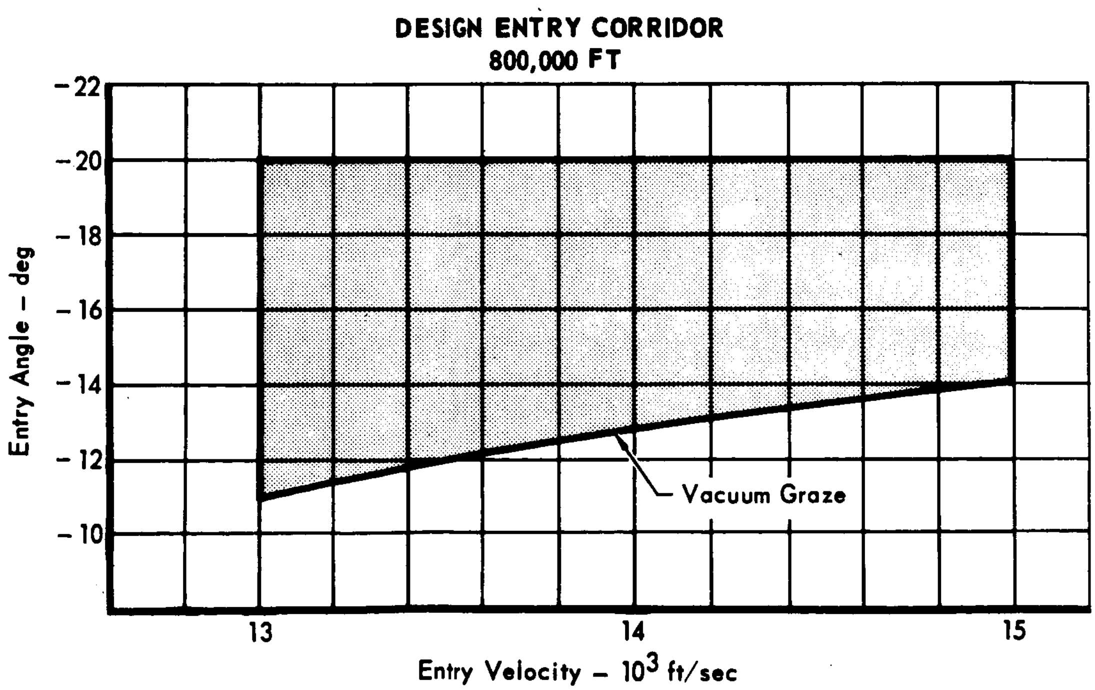


Figure 3–2

Fig. 3-3

OPERATIONAL CONSTRAINTS

Launch:
- Saturn V
- Kennedy Space Flight Center
- 1973 Launch Opportunity
- 2 Identical Planetary Vehicles

Orbit:
- Out-of-Orbit Capsule Landing
- 3 to 12 Days Orbit Stay Time
- Relative Velocity Within ± .2 m/s
- 300 m Minimum Separation Distance at De-orbit Ignition

De-orbit:
- 30° Variation in Landing Location

Entry:
- Altitude = 800,000 ft
- Ballistic (Body of Revolution with c.g. on Centerline)
- Terminal Rocket Deceleration

Landing:
- Vertical Velocity < 25 ft/sec
- Horizontal Velocity < 10 ft/sec

Site:
- 15 to 30° from Terminator
- Spacecraft Pictures Within 600 km of Landing Site, Similar Lighting Conditions as Descent
- TV During Descent
- TV After Landing
- Maximum Data Before Nightfall
- ± 34° Surface Slope

SL Lifetime:
- One Mars Diurnal Cycle Plus Time to Transmit Data

Figure 3–3

ATMOSPHERIC PRESSURE
MODELS VM-1 THROUGH VM-10

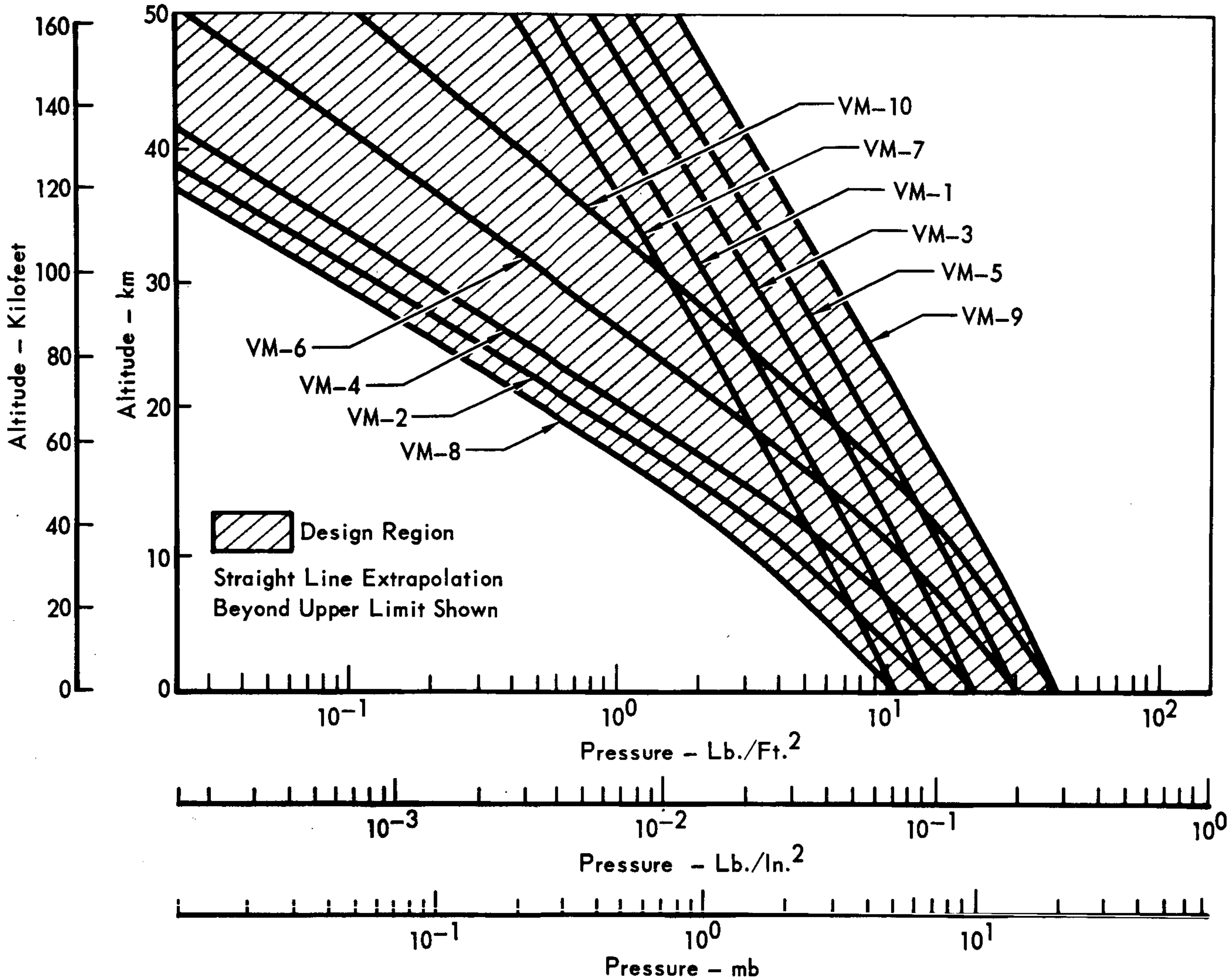


Figure 3–4

SURFACE ENVIRONMENT

Continuous Slopes deg	± 34
*Abrupt Slope Changes deg	± 68
Bearing Capacity psi	6 to ∞
Friction Coefficient	.3 to 1.0
Surface Rocks in	5.0
Length of Surface Slope	324 ft. for 34 deg. slope 6480 ft. for 10 deg. slope
Temperature	+120°F to -190°F
Winds	220 FPS at Alt. of 3.24 ft.

*Local slopes shall not exceed ± 34 deg relative to the horizontal

Figure 3–5

STRUCTURAL DESIGN AND LOAD FACTORS

DESIGN FACTORS

FACTORS OF SAFETY	
• Flight Conditions	1.25
• Ground Handling Conditions Potentially Hazardous to Personnel	1.50
• Emergencies in Air Transport Landing (MIL–A–8421B)	1.00
• Landing System Structure for Mars Landing Condition	1.00

TEMPERATURE FACTORS

Radiative Structures

- Predicted Temperature = Temperature determined from dispersed trajectories.
- Uncertainty Factor = 1.15
- Design Temperature = Initial Entry Temperature + (1.15 x Predicted Temperature Rise)

Ablative Structure

- Predicted Temperature = Temperature determined from dispersed trajectories.
- Uncertainty Factor = 1.25
- Design Temperature = Initial Entry Temperature + (1.25 x Predicted Temperature Rise)

PRESSURIZATION FACTORS

Operating	PROOF	BURST
• Pressurized Compartments	1.33	1.67
• Pneumatic Vessels	1.67	2.22
• Hydraulic Vessels	1.50	2.50
• Lines and Fittings	2.0	4.0
Sterilization (1)		
• Pressurized Compartments	1.05	1.25
• Pneumatic Vessels	1.25	1.50
• Hydraulic Vessels	1.25	1.50
• Lines and Fittings	1.67	2.40

Notes:
(1) Sterilization factors shall be applied to the pressure resulting from the heat of the sterilization cycle or solar heating during the pre-launch phase, whichever is more critical. The pressure shall include the effects of temperature rise, vapor pressure, and other chemical reactions of the enclosed gas or fluid that occur during the cycle.

Figure 3–6

SUMMARY OF RIGID BODY LOAD FACTORS AT THE FLIGHT CAPSULE C.G.

MISSION PHASE	LIMIT LOAD FACTORS (EARTH g's)		REMARKS
	LONGITUDINAL	LATERAL	
ınd oisting	±2.0		Applied independently along hoisting axis; pull-off angles up to 20 degrees.
ssembly	–	±1.2	Cantilevered condition with 360 degree roll capability.
ransportation-air	±3.0	±1.5	Aircraft axis reference; vertical L.F. = ±3.0; not simultaneously
-launch	±2.0		Hoisting – Remarks same as Ground Phase
ınch .ift-off lax Dynamic Pressure -IC End Boost -IC Thrust Decay & Separation	 2.1 2.0 4.9 −1.9	 ±.65 ±.30 ±.10 ±.10	All load factors to be multiplied by 1.2 for dynamic effects.
ection	1.5	±.25	S–IV B Second Burn
erplanetary Cruise	1.0	±.25	Mars Orbit Insertion
osule De-orbit	1.1	Nil	De-orbit Propulsion
osule Entry $\alpha = 0$ $\alpha = 20°$	−21.5 −19.4	0 ±2.2	Maximum Dynamic Pressure Condition
psule Terminal celeration Parachute Terminal Propulsion	 −5.5 −2.5	 −3.9 Nil	
nding	−10.0 −14.0	±10.0 0	Applied simultaneously at Lander C.G.

SECTION 4

PREFERRED CAPSULE DESIGN

A wide variety of candidate concepts was studied to determine a capsule design which meets all of the constraints and will perform the capsule mission successfully. The hazards imposed by the sterilization and long lifetime requirements and by the uncertain environment have necessitated a conservative approach, utilizing redundancy, design margin, and operating flexibility. Mission profile studies were used to determine the range of profiles which satisfy mission objectives and environmental constraints and which are compatible with the capabilities of the Flight Capsule and other VOYAGER systems.

A continually evolving baseline configuration was used as the basis of the studies, in order to permit concurrent work on all elements of the system. This baseline identified the requirements on the various subsystems during each mission phase, the alternative methods of satisfying the requirements, and a continually updated preferred selection from among these alternatives.

We sought optimization of the entire system, rather than any individual subsystem. Decisions so important that they influence the basic characteristics of the system were made as the result of major trade studies and system analyses. Probability of mission success was the most important optimization criterion; others were system performance, development risk, versatility, and cost.

Deceleration from entry velocity to landing on the Martian surface was the subject of several Capsule Bus trade studies. Selection of the aerodynamic and propulsive subsystems which perform the descent and terminal deceleration have a major effect on other flight equipment. Choice of a lander configuration which will operate satisfactorily in the surface environment significantly affects installation of the other Capsule Bus subsystems, as well as the Surface Laboratory. Selection of the thermal control subsystem was one of the more important studies for the Surface Laboratory, since it had a major influence on configuration, power subsystem, and the mission profile. Of comparable importance were the optimization studies for the telecommunications subsystem and the installation trade-offs of the science instruments. For the Entry Science Package, the more critical trade studies were those leading to optimization of the interfaces between the science instruments and the Aeroshell.

The effects of modifying the constraints have been evaluated parametrically but the preferred configuration was selected to satisfy all of the imposed constraints.

The Flight Capsule described by our preferred design, Figure 4-1, has a gross weight of 5000 pounds of which 3680 pounds is assigned to the Capsule Bus, 180 pounds to the Entry Science Package, 916 pounds to the Surface Laboratory, and 224 pounds as a weight margin. Figure 4-2 is a weight summary, listing weights of major capsule elements at several significant points in the mission profile. The weight listed for each element contains redundancies incorporated to improve the probability of mission success. The redundant items weight 73 pounds.

Incorporation of redundancy was guided by use of a mission effectiveness analysis which identified the priority for allocating weight for this purpose. Adding the redundant items increased the total reliability of the Flight Capsule (less experiments) from .46 to .71. It increased the probability of achieving at least one of three primary mission objectives - landing, entry science, landed science - from .80 to .90, see Figure 4-3. These three mission goals - achievement of a Flight Capsule landing, performance of entry science experiments and performance of landed science experiments - were assigned relative values of 0.40, 0.35, and 0.25, respectively. This order of priority was established by the VOYAGER Mission General Specification.

FLIGHT CAPSULE PREFERRED DESIGN

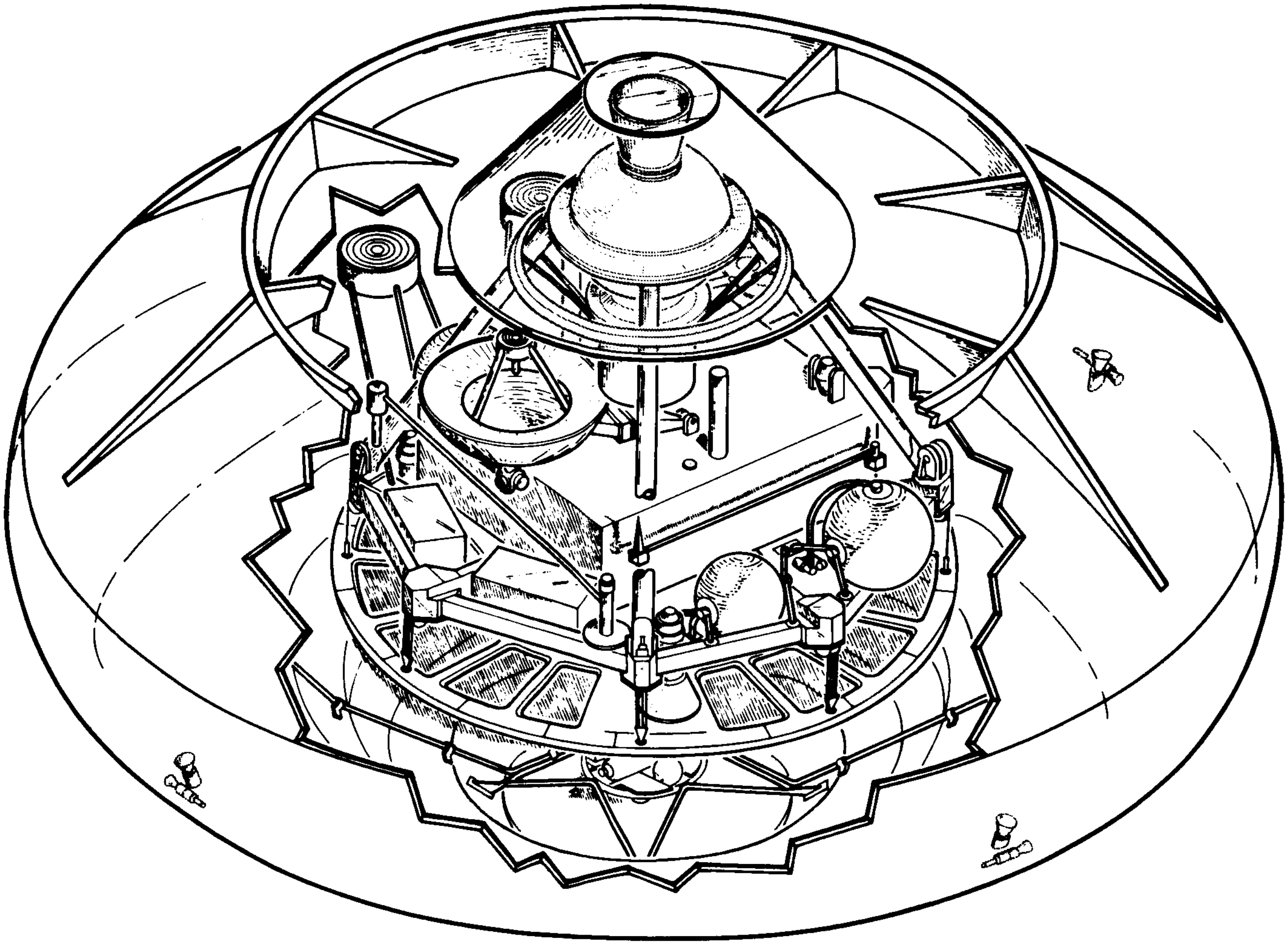


Figure 4–1

FLIGHT CAPSULE WEIGHT SUMMARY

	LAUNCH WEIGHT	DEORBIT PROPULSION INITIATION WEIGHT	ENTRY WEIGHT	TERMINAL PROPULSION INITIATION WEIGHT	TOUCHDOWN WEIGHT
CAPSULE BUS	(3680)	(2923)	(2415)	(1540)	(1321)
Sterilization Canister & Adapter	735				
Aeroshell	642	622	618		
Lander	2303	2301	1797	1540	1321
SURFACE LABORATORY	(916)	(916)	(916)	(916)	(916)
Science Experiments	110	110	110	110	110
Supporting Equipment	806	806	806	806	806
ENTRY SCIENCE PACKAGE	(180)	(180)	(180)	(178)	(178)
Science Experiments	27	27	27	25	25
Supporting Equipment	153	153	153	153	153
TOTAL FLIGHT CAPSULE	4776	4019	3511	2634	2415
WEIGHT MARGIN	224				
FLIGHT CAPSULE (Maximum Weight)	5000				

Figure 4-2

FLIGHT CAPSULE RELIABILITY

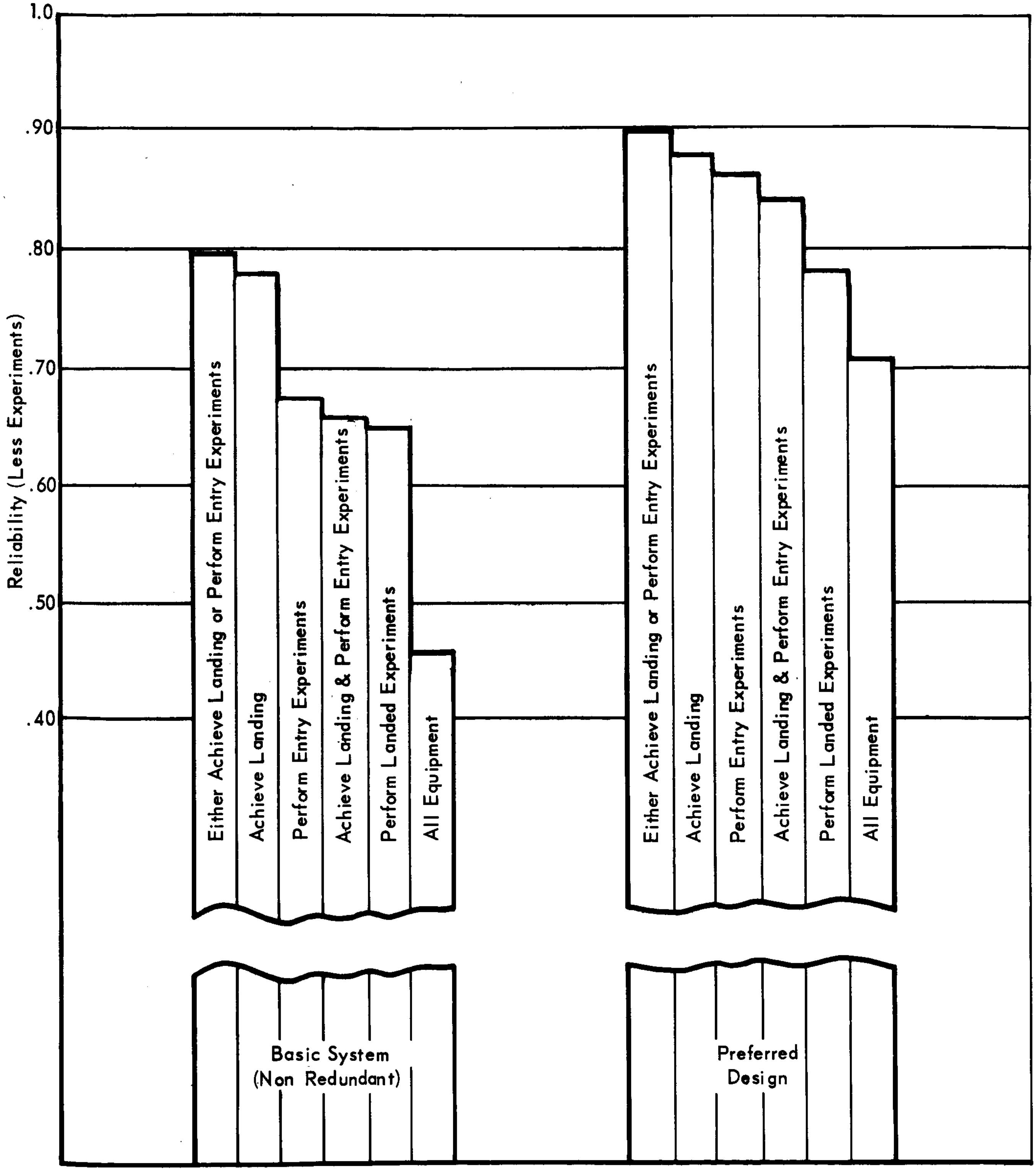


Figure 4-3

4.1 MISSION ANALYSIS - Studies of the Flight Capsule mission profile, coupled with the specification of the mission objectives, have led to the determination of specific functional requirements for the Flight Capsule systems and subsystems.

Our mission analysis included a study of the available Martian orbits, since these determine the initial conditions for a major portion of the capsule mission. Similarly, we established a range of landing sites which must be attainable by the Flight Capsule, to define the end conditions of the profile. The major portion of our mission analysis has then been devoted to the transfer from initial to end conditions.

Both a nominal mission profile plus deviations from the nominal, to encompass a wide range of possible operating conditions, must be established. Accordingly, we have designed for operation at any point within a design requirements range, rather than at one specific design performance point. This provides a triple benefit.

a. First, it fosters concurrent analysis of mutually dependent aspects of the design.

b. Second, it reserves for mission operations planners the flexibility in mission selection that is necessary to account for factors not yet well defined, such as instrument selection, precise subsystem characteristics, and latest environment data.

c. Finally, since this approach usually imposes more rigorous requirements than designing to a single performance point, it provides a margin of conservatism appropriate to the present maturity of interplanetary exploration.

4.1.1 Landing Site Constraints - A range of suitable landing sites has been postulated on the basis of attaining:

a. Satisfactory surface lighting at the landing site (15 to 30 degrees to the terminator).

b. Close examination of regions with seasonal color change (within 10° latitude N and 40° latitude S).

c. Maximum data transmission prior to the onset of Martian night.

4.1.2 Planetary Orbit and Deorbit Considerations - The 1973 mission includes two Planetary Vehicles - each consisting of a Flight Spacecraft and a Flight Capsule. The two vehicles are launched on one Saturn V, but their arrival times near Mars are staggered by about eight days, in order to eliminate potential communications interference and permit efficient usage of the Deep Space Net. The operational considerations discussed below are equally applicable to both Flight Capsules.

Planetary orbits with maximum apoapsis altitudes of 20,000 km and minimum periapsis altitudes of 700 km have been studied. The minimum periapsis was established by orbit lifetime, to meet planetary quarantine restrictions. Orientation of these orbits for optimum landing conditions influences the launch window and required orbital insertion velocity supplied by the spacecraft. From operational and design considerations, we prefer a near morning terminator landing even though this imposes somewhat more stringent launch and insertion velocity requirements. This is shown in Figure 4.1-1.

The Flight Capsule de-orbit profile depends on the magnitude and direction of the de-orbit velocity increment and its point of application (de-orbit anomaly). From the range of anomalies and velocity increments which satisfy requirements for a sufficiently broad entry corridor and line-of-sight communication with the Spacecraft, those that maximize Flight Capsule performance were selected. Maximizing performance requires:

a. Reducing delivery system weight by limiting the de-orbit velocity increment, the de-orbital descent time, and atmospheric pressure and thermal loads.

b. Maximizing the landing site selection flexibility.

c. Reducing landing site dispersions.

d. Retaining operational flexibility.

Based on these considerations, we have chosen a de-orbit velocity increment of up to 950 ft/sec, which satisfies the constraints and provides a large degree of entry flexibility. The resultant entry corridor is shown in Figure 4.1-2 and includes entry velocities of 13,000 to 15,000 ft/sec and flight path angles from vacuum graze to -20°. The shaded area in the figure shows that portion of the entry corridor available for a 1,000-10,000 km orbit when applying a de-orbit velocity increment from 600-950 ft/sec tangentially to the orbit path.

For line-of-sight communication between the entering Flight Capsule and the orbiting spacecraft, constraints must be placed on the de-orbit anomaly. This is shown in Figure 4.1-3 which presents the permissible boundary between de-orbit anomaly, Θ_D, and the location of the periapsis with respect to the terminator, ε. The boundary applies to a landing near the morning terminator.

LANDING TIME VS LAUNCH PERIOD AND MARS INSERTION VELOCITY

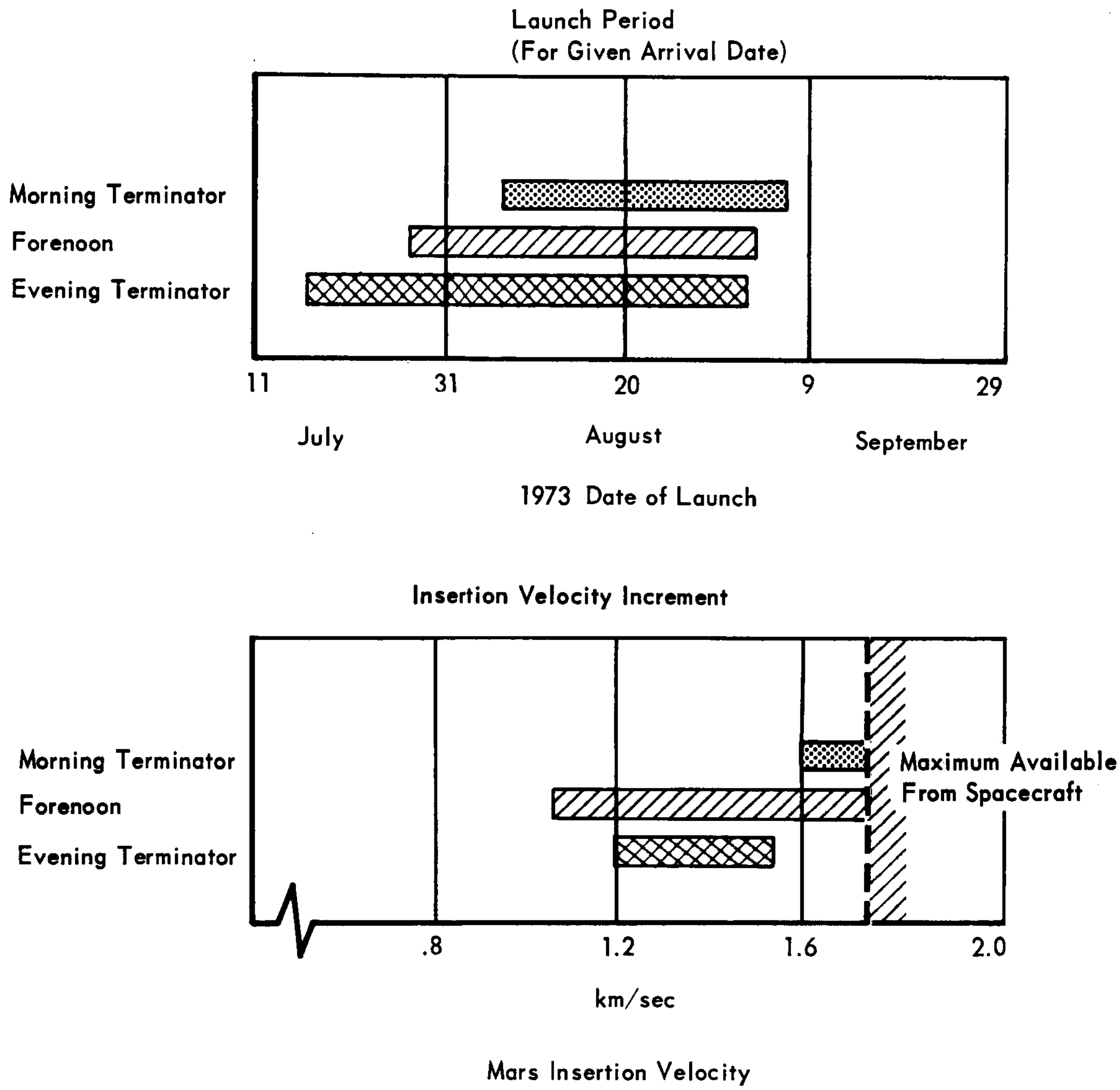


Figure 4.1-1

ENTRY CORRIDOR

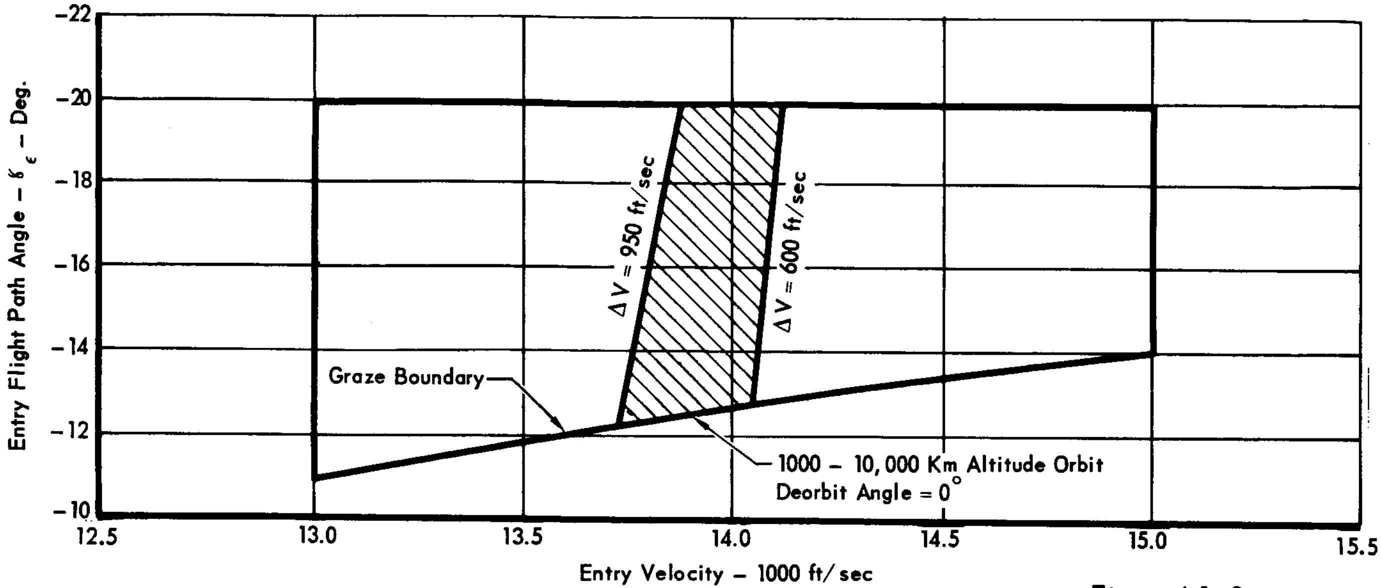


Figure 4.1-2

ALLOWABLE DE-ORBIT ANOMALY

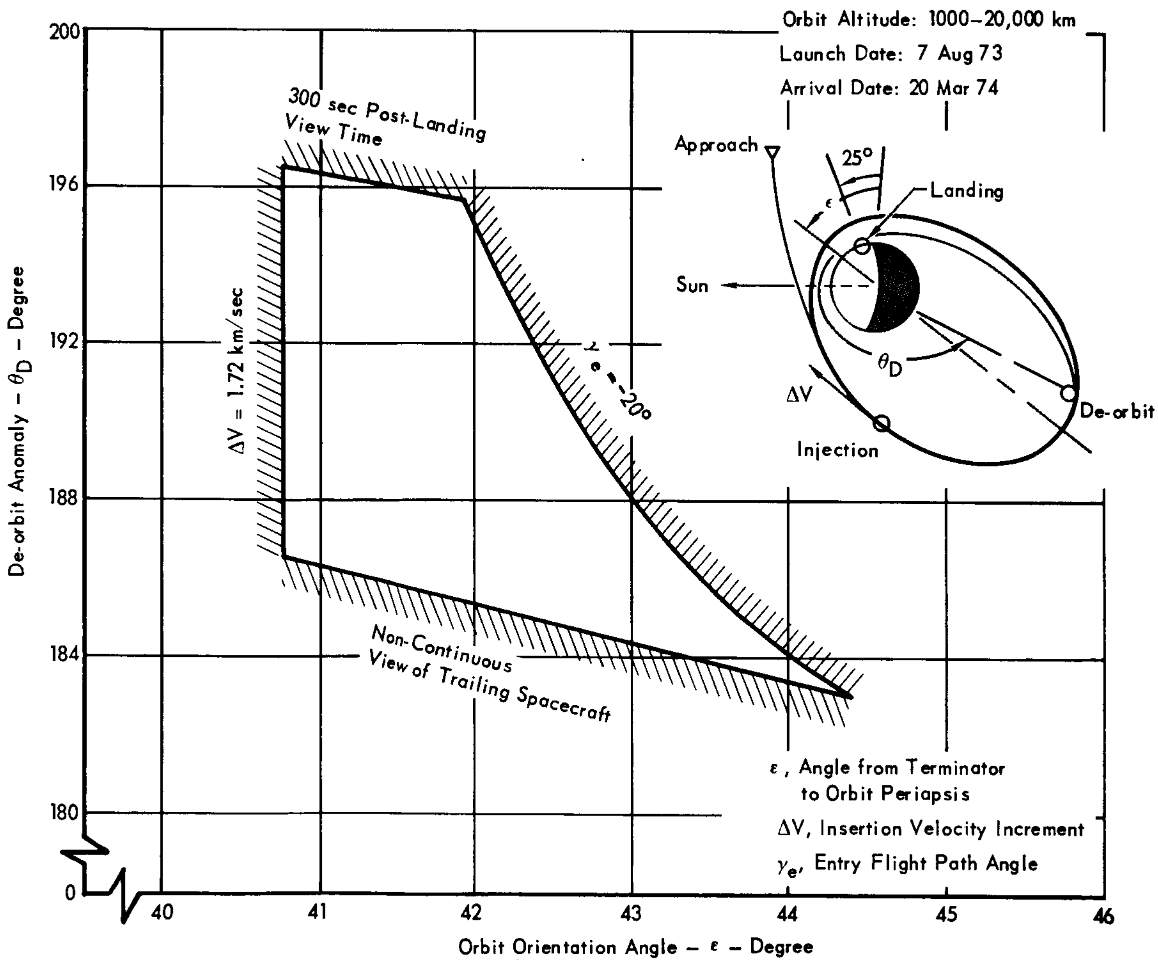
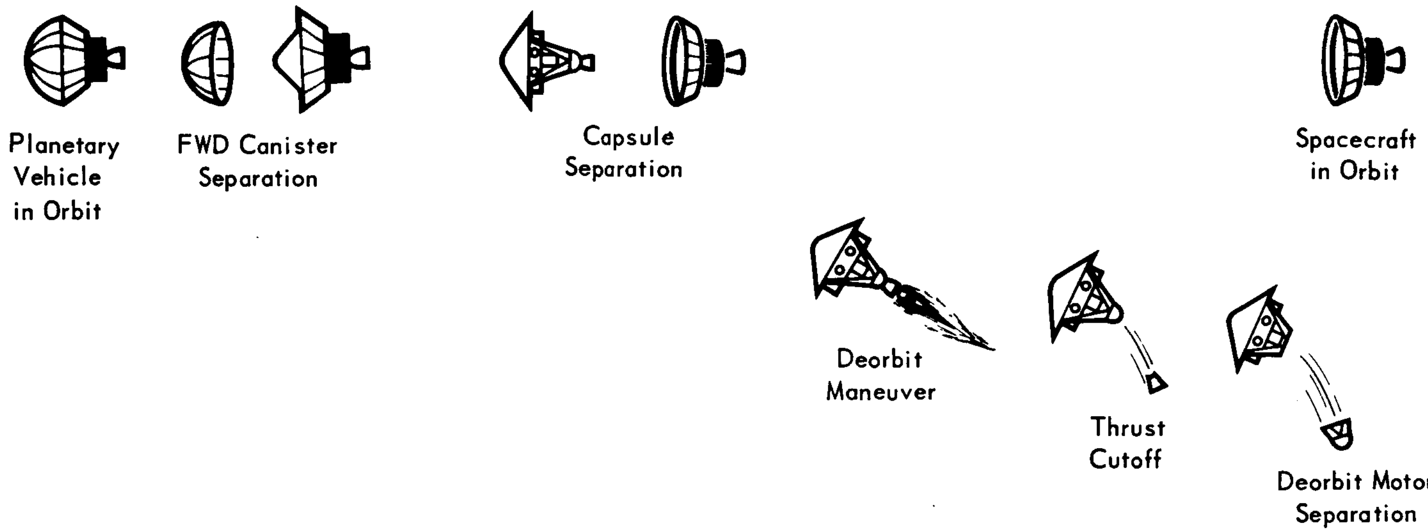


Figure 4.1-3

4.1.3 Mission Profile - Figure 4.1-4 shows the nominal Flight Capsule mission, starting with launch and concluding with surface operation on Mars. Of special significance are the mission steps subsequent to entry. At an altitude of 23,000 ft, a supersonic parachute is deployed. Twelve seconds later, the Aeroshell is released. When the Capsule Lander descends to 5,000 ft, four terminal propulsion engines are ignited, at 50% maximum thrust. The lander is then released from its parachute. At about 0.5 sec after ignition, the terminal propulsion engines are throttled to provide a constant 0.8 g deceleration level until a pre-programmed descent profile is intersected. Attitude control is maintained by differential throttling of the terminal propulsion engines. When the lander descends to an altitude of 50 ft, the terminal propulsion unit provides a constant-velocity descent of 5 ft/sec. This constant descent velocity is maintained to 10 ft above the Martian surface, where the terminal propulsion subsystem is shut down. The Capsule then falls free and lands on the surface of Mars. The Surface Laboratory will operate to perform the landed science experiments for at least one diurnal cycle.

Characteristics of a typical trajectory, which is well within the design envelope and which satisfies the constraints, are summarized in Figure 4.1-5.

MAJOR VOYAGER MISSION EVENTS



PLANETARY VEHICLE MISSION

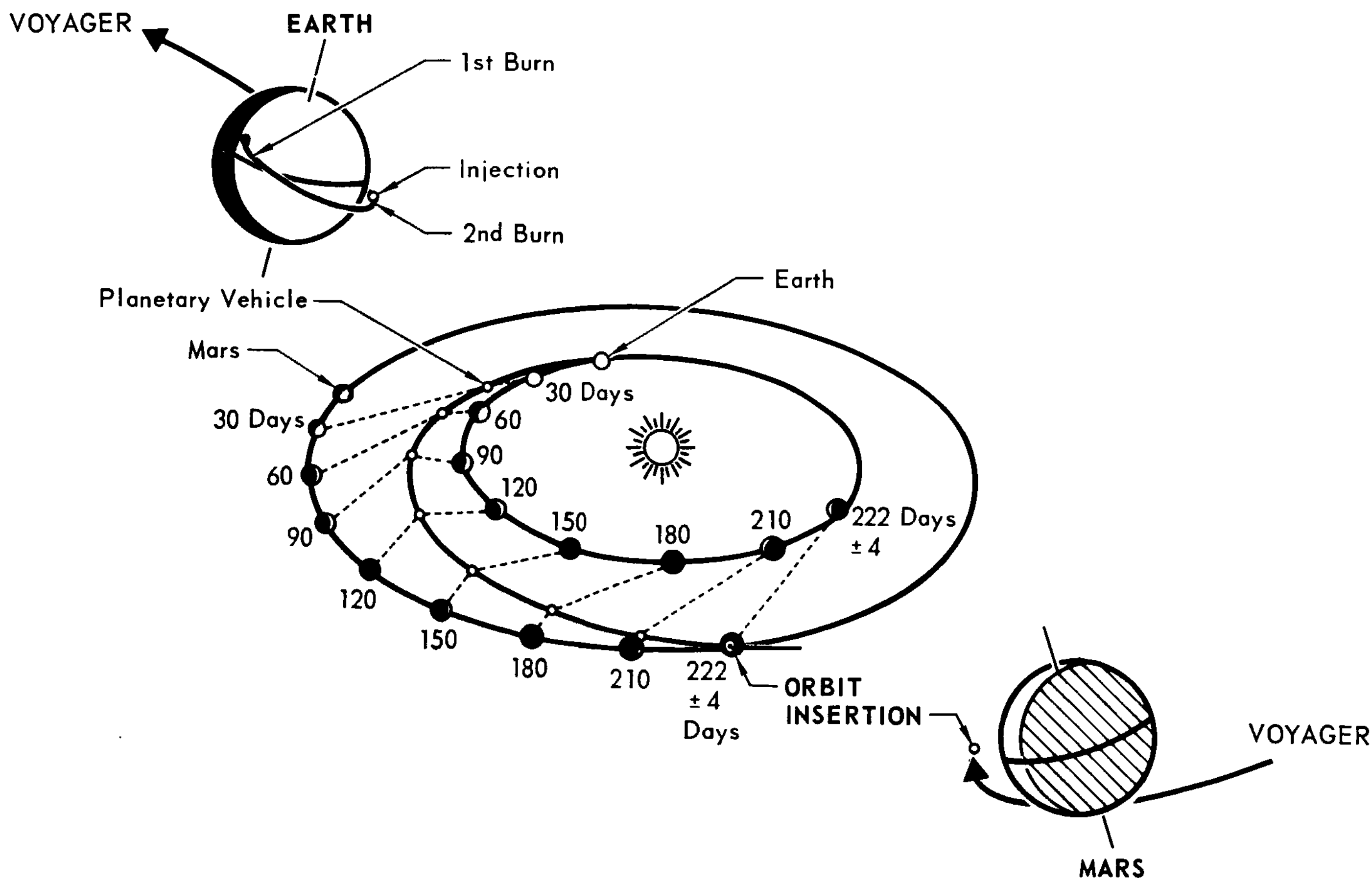
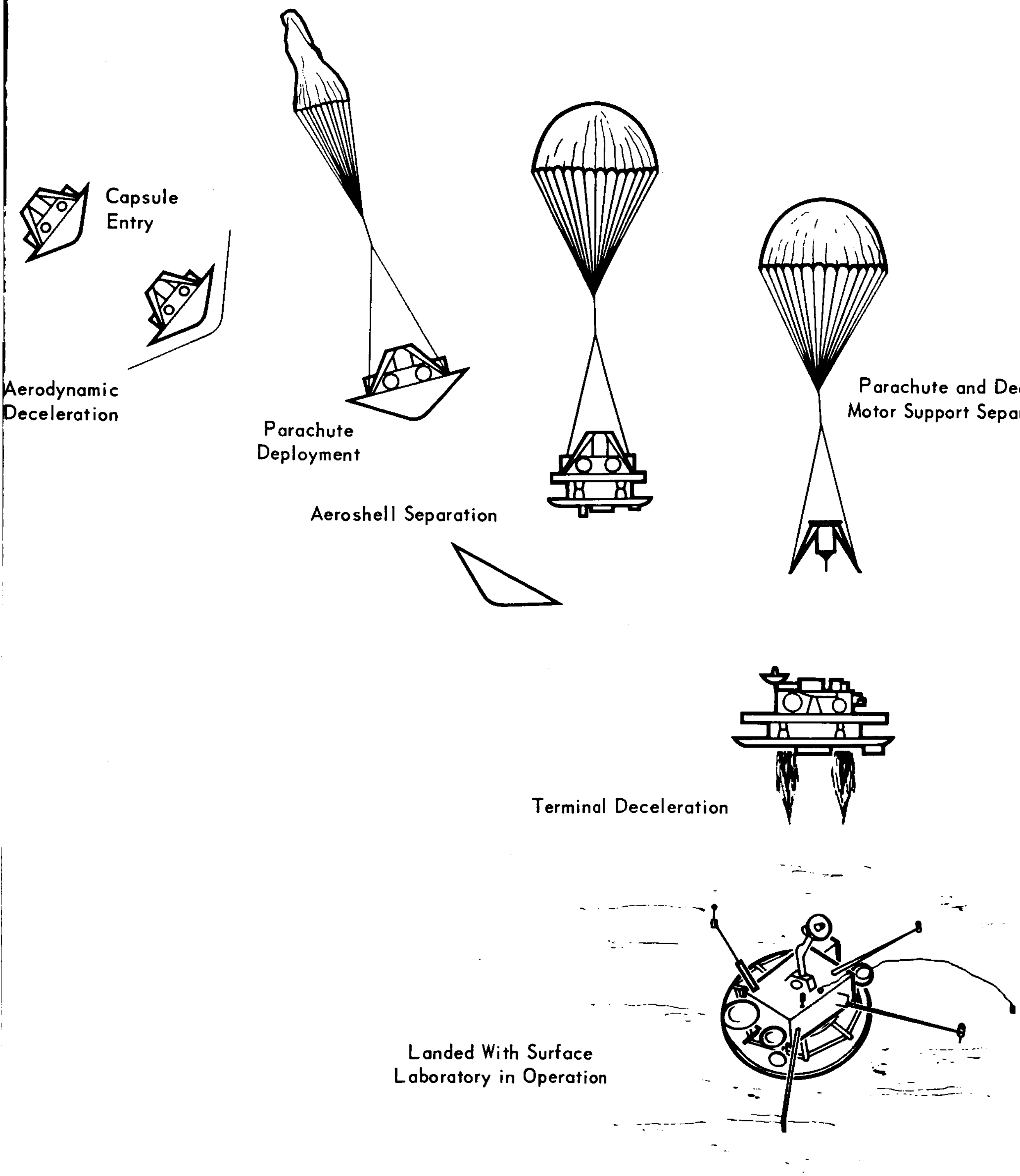


Figure 4.1-4


Capsule
Entry
Aerodynamic
Deceleration
Parachute
Deployment
Aeroshell Separation
Parachute and Deorbit
Motor Support Separation
Terminal Deceleration
Landed With Surface
Laboratory in Operation

1973 TYPICAL MISSION PROFILE SUMMARY

MISSION PHASE	CONSTRAINT	TYPICAL
Launch-Injection-Interplanetary		
Launch Site	KSC, Complex 39	KSC, Complex 39
Vehicle	Saturn V with 2 PV	Saturn V with 2 PV
Date	13 July 1973 to 6 Sept. 1973	7 August 1973
Period	30 Days	30 Days Remain
Azimuth	90 to 115 degs	115 deg (113.6 deg min)
Firing Window	$\geq$ 1 hour	1.05 hours
Parking Orbit	10 to 90 min	32.6 min
Injection Gross Weight	55,300 lbs	55,300 lbs
Vis Viva Integral	$\leq$32.5 km^2/sec^2	16.7 km^2/sec^2
Declination, Outgoing	5 to 36 degs	36 deg
Interplanetary Cruise Trajectory	Type I	Type I (153.8 deg)
Transit Time	157 to 224 days	222 + 4 days
Inclination to Ecliptic	> 0.1 deg	3.6 degs
Arrival Date Separation	> 8 days	8 days
Maneuver Timing	2 to 20 days	6.8 days
Velocity Increment	< (210 – 10) m/sec	168 m/sec
Orbit Insertion-Operations		
Arrival Date	3 Feb. 1974 to 21 Mar. 1974	20 Mar. 1974
Separation from Nominal	(± 4 days)	± 4 days
Hyperbolic Excess Speed	$\leq$3.25 km/sec	2.58 km/sec
Orbit Insertion Maneuver	—	Tangent Method
Velocity Increment, Impulsive	1.76–0.04 km/sec	1.69 km/sec
Apsides Rotation Angle	> ± 20 deg	–61.6
View from Goldstone DSIF	Required	15 deg Above Horizon
Orbit	Elliptical	Elliptical
Inclination to Equator	$\geq$ 30 deg	40 deg
Periapsis Altitude	500 to 1,500 km	1,000 km
Apoapsis Altitude	10,000 to 20,000 km	20,000 km
Periapsis Location, Initial	Near Either Terminator	Near Morning Terminator
First 90 days	45 to 0 degs to Nearest Terminator	42.6 to 5.7 degs to Morning Terminator
Next 90 days	90 to –30 degs to Nearest Terminator	5.7 to –31.1 degs to Morning Terminator
Latitude	40°N to 60°S	20.47°S
Occultation, Sun by Mars	None for 30 days	None for 30 days
Canopus by Mars	None for 30 days	None for 30 days
Earth by Mars	< (%) Orbit Period	None for 30 days
De-Orbit-Descent-Entry-Decelerator		
De-Orbit Time from Insertion	3 to 12 days	3.1 days
Anomaly	—	187 deg
View from Goldstone DSIF	Earth Required	55 deg Above Horizon
Entry	800,000 ft	800,000 ft
Flight Path Angle	–20 deg to Graze	–19.0 deg
View from Goldstone DSIF	Earth Required	53 deg Above Horizon
Body	—	120 deg Sphere-Cone
Ballistic Coefficient	—	.266 Slugs/ft^2
Atmospheric Model	VM–1 to VM–10	VM–9
Aero-Decelerator	—	Parachute
Altitude	—	23,000 ft
Terminal Propulsion	—	LPR (4 Engine)
Altitude	—	5,000 feet
Landing-Post Landing Operations		
Landing Site	Not Yet Defined	Syrtis Major
Latitude	10°N to 40°S	0°
Vertical Velocity	< 25 ft/sec	16 ft/sec
Lighting Angle	15 to 30 deg to Terminator	25 deg to Morning Terminator
View, Spacecraft from Capsule	$\geq$ 34 deg Above Horizon	60 deg Above Horizon
View, Earth from Capsule	$\geq$ 34 deg Above Horizon	57 deg Above Horizon
View, Site from Earth	Earth Desired	49 deg Above Horizon
Post Landing Daylight	Maximum Data Before Night	10.5 hours Before Night
Communications with Spacecraft	Confirmation of Landing	9 min
Communications with Earth	Maximum Transmission First Day	5.8 hours (2.9 hours with Goldstone)

Figure 4.1–5

4.2 CAPSULE BUS SYSTEM - The Capsule Bus, after separation from the orbiting spacecraft, delivers the Entry Science Package into the Martian atmosphere and the Surface Laboratory to the surface of Mars. Our preferred design has the following basic features:

a. Simple, conservative, state-of-the-art approaches are used wherever possible.
b. To the extent practicable, the Capsule Bus (CB), the Surface Laboratory (SL), and the Entry Science Package (ESP), are independent and separable modules.
c. The physical interface between the Capsule Bus and the Surface Laboratory is a structural field joint and a single electrical connector. The interface with the Entry Science Package is more complicated, because of the sensor attachments needed at selected places of the Aeroshell. However, all the ESP support equipment is contained within one module, having a simple interface with the Capsule Bus.
d. Flight Capsule equipment designed to enter the Martian atmosphere is biologically sealed in a Sterilization Canister. It is terminally sterilized by heating in dry nitrogen, and remains sealed until it is separated in flight.
e. The Capsule Bus is designed for soft landing in a controlled, upright attitude. Soft landing for '73 implies an impact acceleration of 14 g; however, all equipment is designed to withstand a design load of 22 g, to accommodate peak entry loads.
f. Wherever practical, basic elements of the Capsule Bus are standardized for future missions. In this regard, the Flight Capsule weight for future missions is 7000 pounds.
g. The Capsule Lander configuration is compatible with a mobile Surface Laboratory for later missions.
h. Demonstrated reliability is a primary design requirement.
i. Spaceflight-proven hardware and approaches are used wherever possible.
j. Single failure modes are eliminated where practical. Exceptions are: single de-orbit motor, single Aeroshell/heat shield, single parachute, and single landing system. In no case, is system reliability unduly threatened.
k. Suitable diagnostic information is collected throughout the mission and transmitted for complete analysis of system performance.

The Capsule Bus consists of three major structural modules: the Sterilization Canister and adapter, the Aeroshell, and the Capsule Lander. These modules house all the Capsule Bus equipment, as well as the Entry Science Package and the Surface Laboratory. Figure 4.2-1 presents our preferred Capsule Bus design and Figure 4.2-2 shows an interior arrangement. Of interest are the various staging sequences from start of Capsule Bus separation from the Spacecraft down to landing and Surface Laboratory deployment. This is shown in Figure 4.2-3.

Figure 4.2-4 presents the weight statement for the preferred design, including all allotments for contingency, standardization, and redundancies. A weight uncertainty of $\pm$ 256 pounds has been calculated for the Capsule Bus, based on statistical variation and estimation techniques. It can be expected that this uncertainty would not be exceeded as long as requirements and criteria are not changed.

4.2.1 Major Structural Modules

4.2.1.1 Sterilization Canister/Adapter - A Sterilization Canister is provided to protect the Capsule Bus from recontamination after terminal sterilization. The canister is of conventional aluminum sheet and stiffener construction and incorporates a field joint at the maximum diameter which also serves to support the dual installation of a contained explosive separation device.

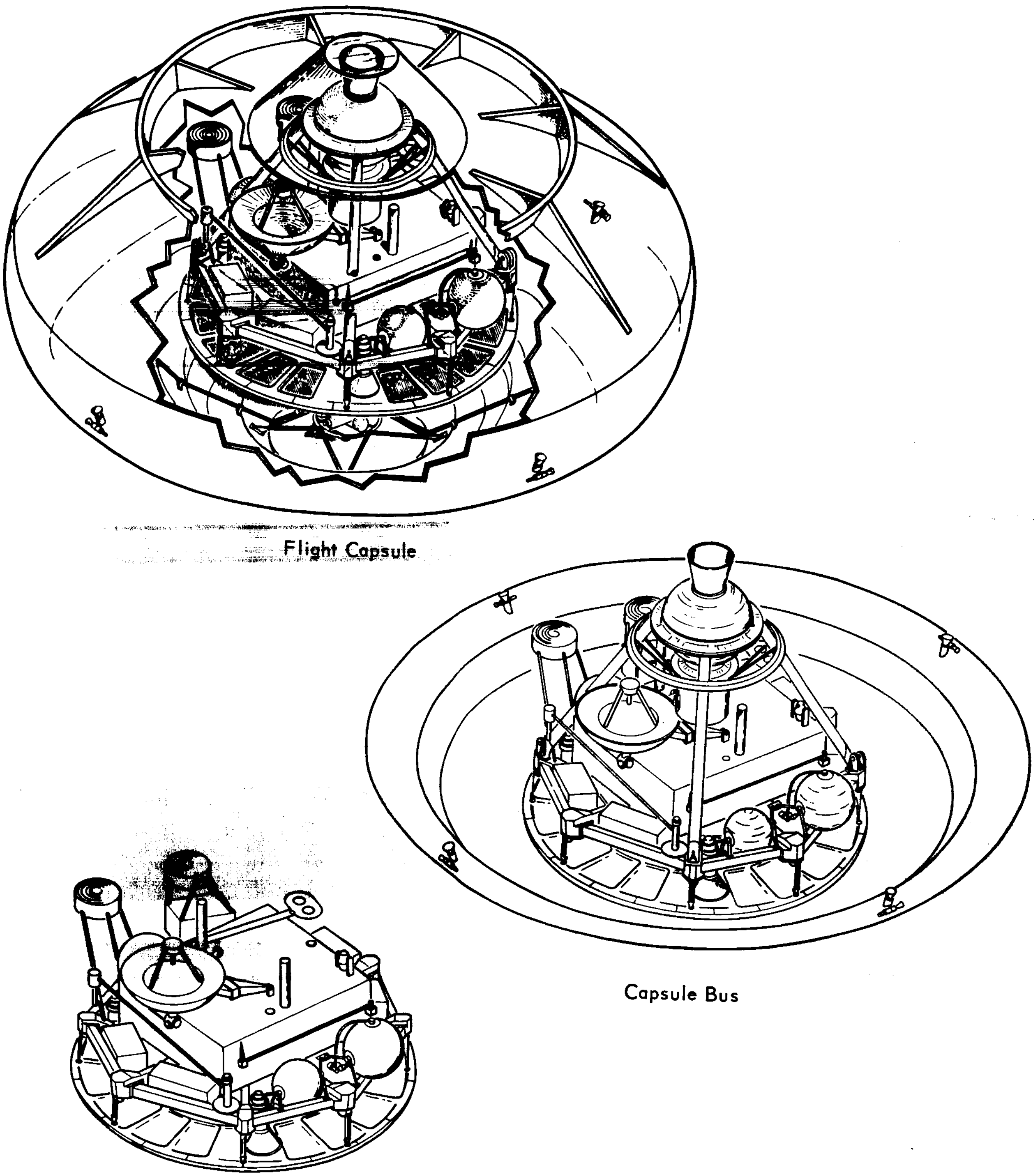
The forward section of the canister is hemispherical to make maximum use of the envelope specified for the Capsule Bus. It is reduced to a smaller radius at the separation plane to minimize discontinuities. The aft section is also hemispherical to attain an efficient pressure vessel and provides the structural connection to the Spacecraft.

The adapter, which attaches the Flight Capsule to the aft canister, is of basic truss construction. Both the aft canister and the adapter are attached to the Spacecraft by eight attachment fittings.

The canister incorporates a pressurization and venting device to allow circulation of gases during terminal sterilization and to permit venting during launch without the danger of recontamination. Figure 4.2-5 highlights the structural arrangement of the canister and adapter within the Capsule Bus, and includes both a description of the preferred design characteristics and a summary chart of the candidate designs which were considered in trade-off analyses.

Operationally, canister separation is as follows. The contained explosive separation devices are fired, shearing 300 drilled titanium retaining bolts. The forward canister is then jettisoned at a velocity of 1.25 ft/sec by the energy of the explosive charge. After the Spacecraft is repositioned, the four explosive bolts which attached the Capsule Bus to the adapter are fired and the Capsule Bus separates

CAPSULE BUS PREFERRED DESIGN



Flight Capsule

Capsule Bus

Lander

Figure 4.2-1

CAPSULE BUS INTERIOR ARRANGEMENT

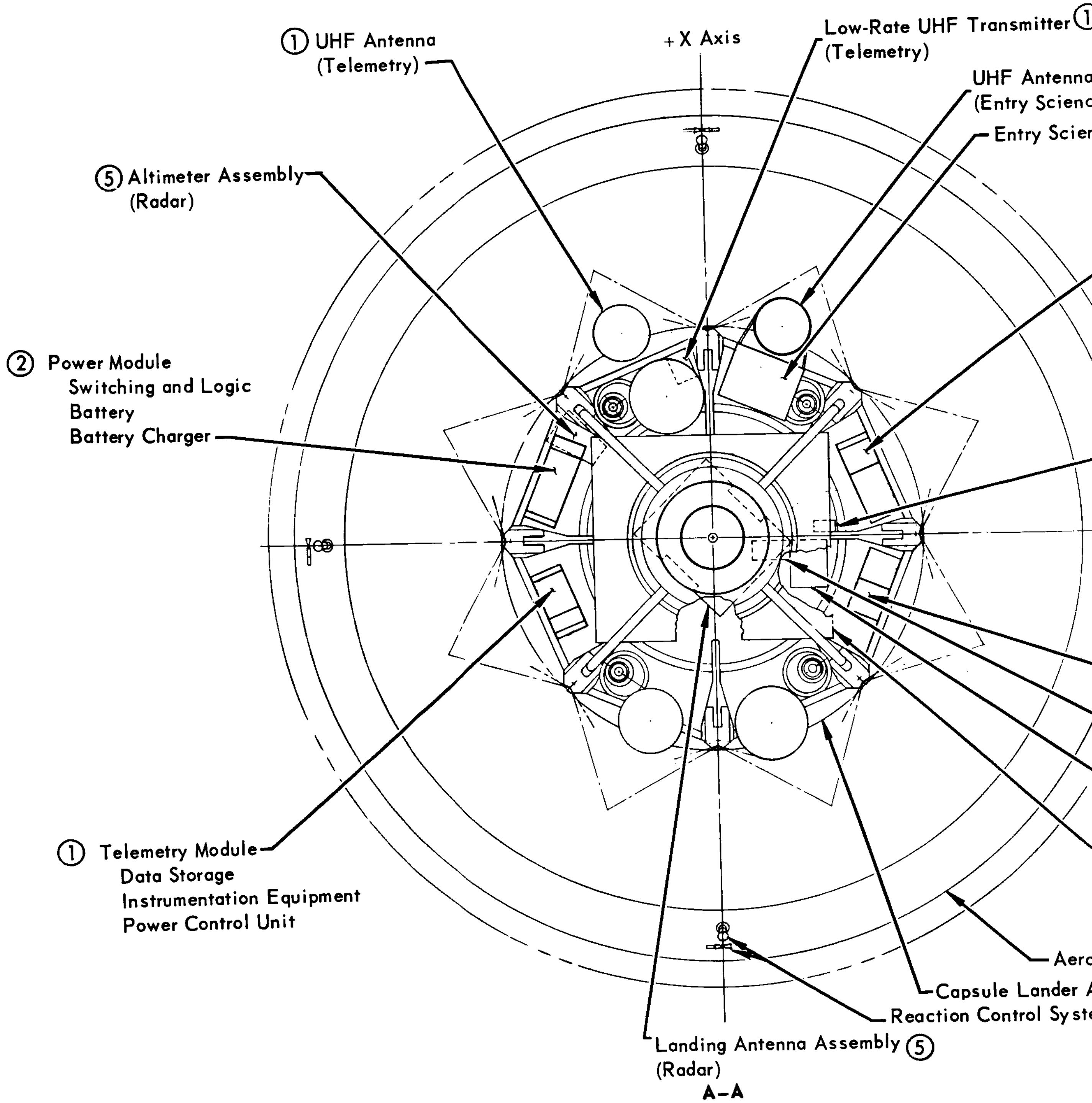
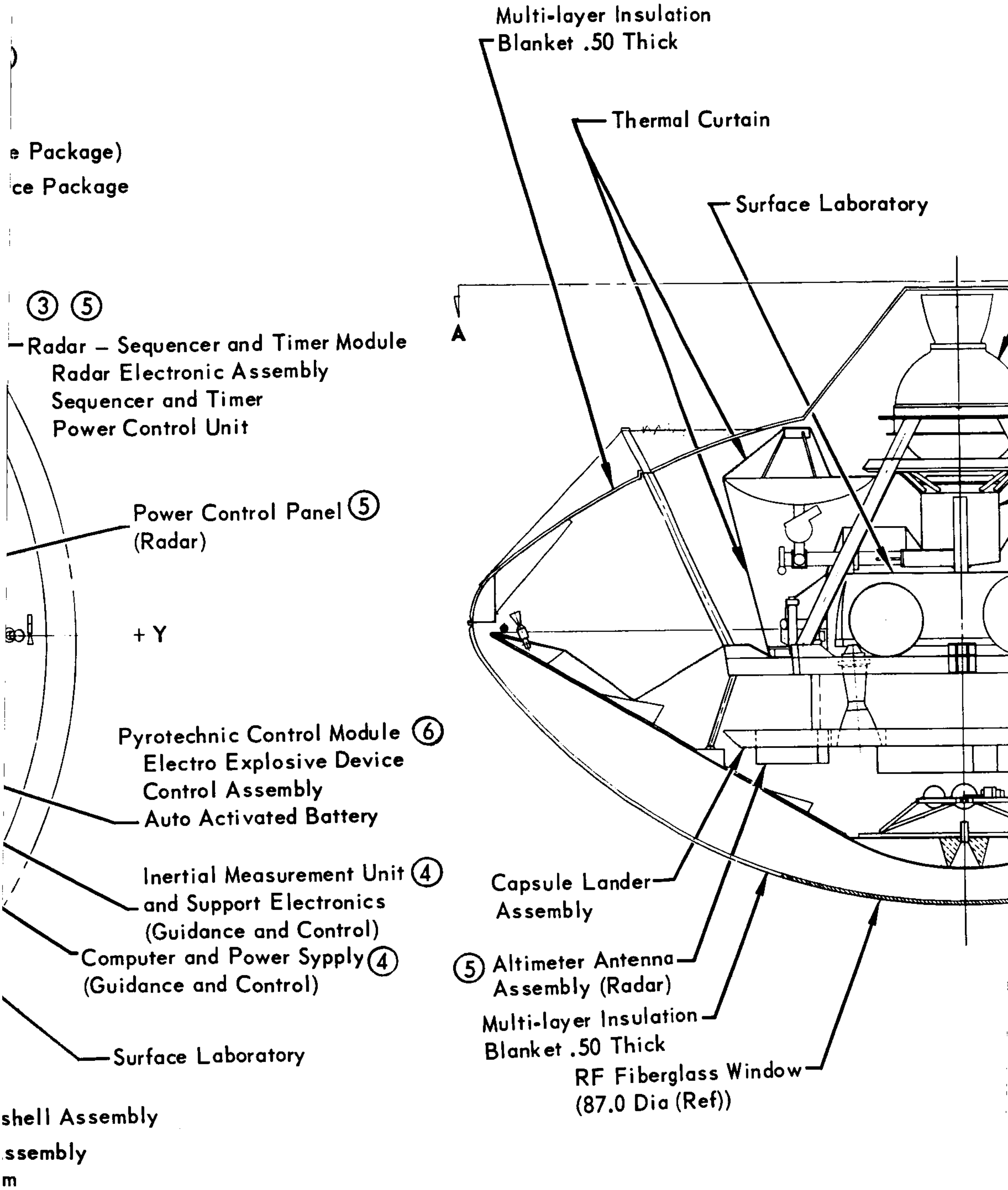
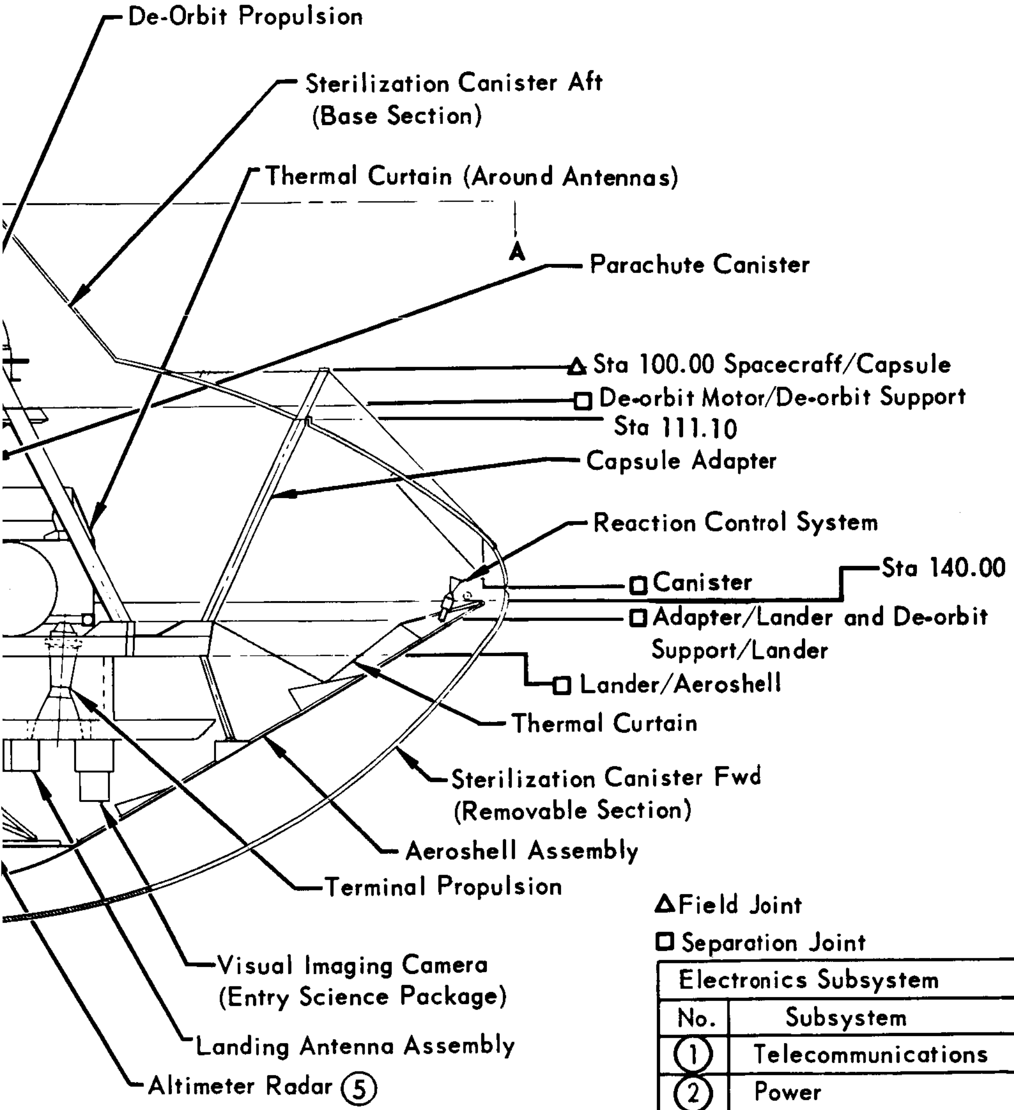


Figure 4.2-2

4-16 -1


e Package)
ce Package
③ ⑤
Radar – Sequencer and Timer Module
Radar Electronic Assembly
Sequencer and Timer
Power Control Unit
Power Control Panel ⑤
(Radar)
+ Y
Pyrotechnic Control Module ⑥
Electro Explosive Device
Control Assembly
Auto Activated Battery
Inertial Measurement Unit ④
and Support Electronics
(Guidance and Control)
Computer and Power Sypply ④
(Guidance and Control)
Surface Laboratory
shell Assembly
ssembly
m
Multi-layer Insulation
Blanket .50 Thick
Thermal Curtain
Surface Laboratory
A
Capsule Lander
Assembly
⑤ Altimeter Antenna
Assembly (Radar)
Multi-layer Insulation
Blanket .50 Thick
RF Fiberglass Window
(87.0 Dia (Ref))

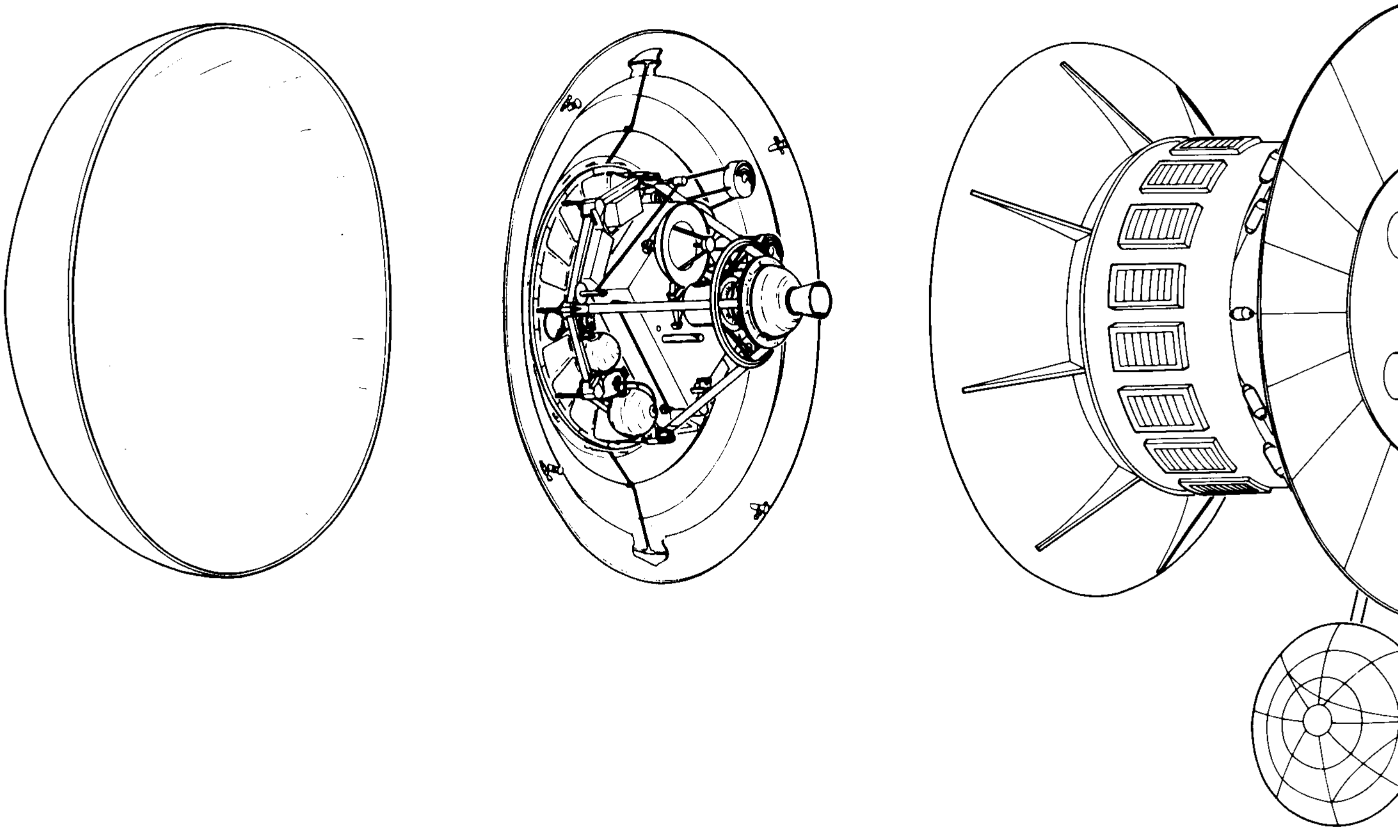


△ Field Joint

□ Separation Joint

Electronics Subsystem	
No.	Subsystem
1	Telecommunications
2	Power
3	Sequencing and Timing
4	Guidance and Control
5	Radar

CAPSULE BUS STAGING SEQUENCE

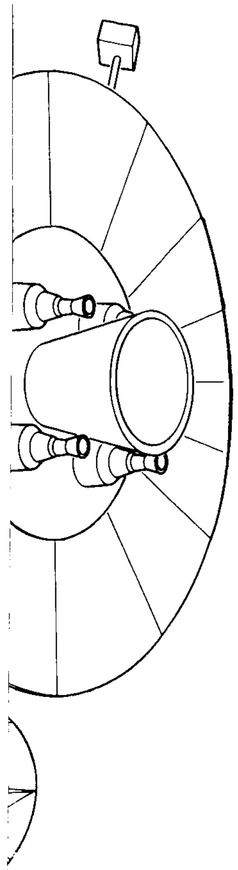


(1) FORWARD CANISTER SEPARATION
- Redundant CESD Severs Canister Attach Bolts in Tension
- Also Provides Energy to Separate Canister
- Separation Velocity > 1.25 Ft/Sec

(2) CAPSULE BUS SEPARATION
- Fire Eight Explosive Bolts at Adapter/ Capsule Lander Interface
- Operate Pitch and Yaw Thrust Chambers to Separate
- Separation Velocity = 1.25 Ft/Sec.

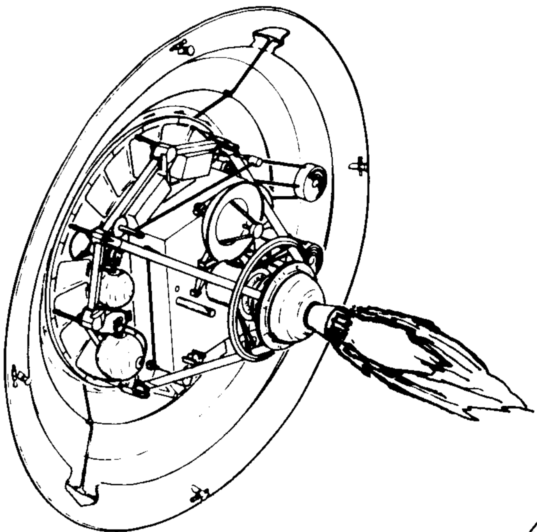
(3) SPACECR ORBIT
- Aft Cani on Space
- Adapter to Aft Ca

Figure 4.2-3

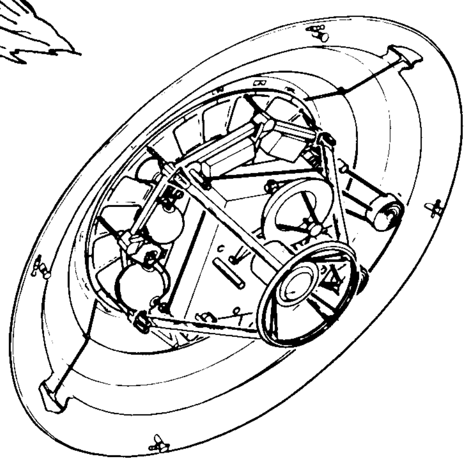


AFT IN

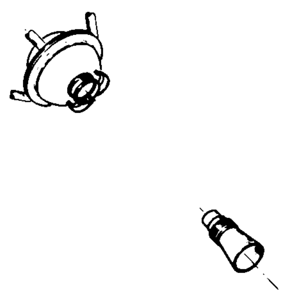
ster Remains
craft
Remains Attached
mister.



(4) DEORBIT MOTOR FIRING

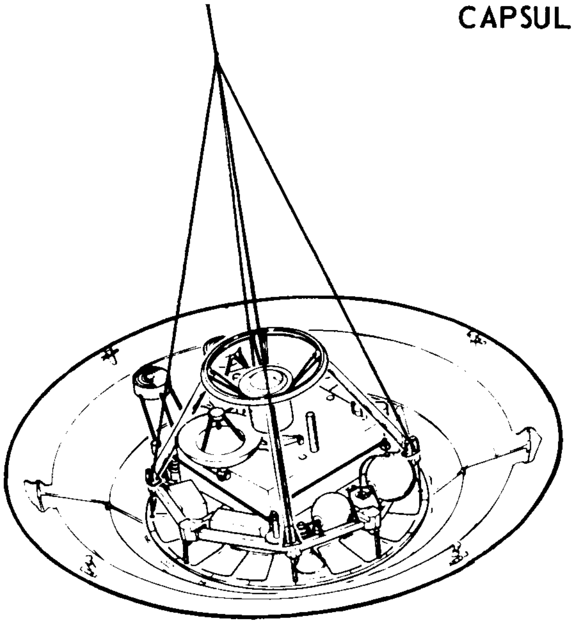


(5) DEORBIT THRUST CUTOFF
- Fire One Explosive Bolts which Releases Nozzle from Case

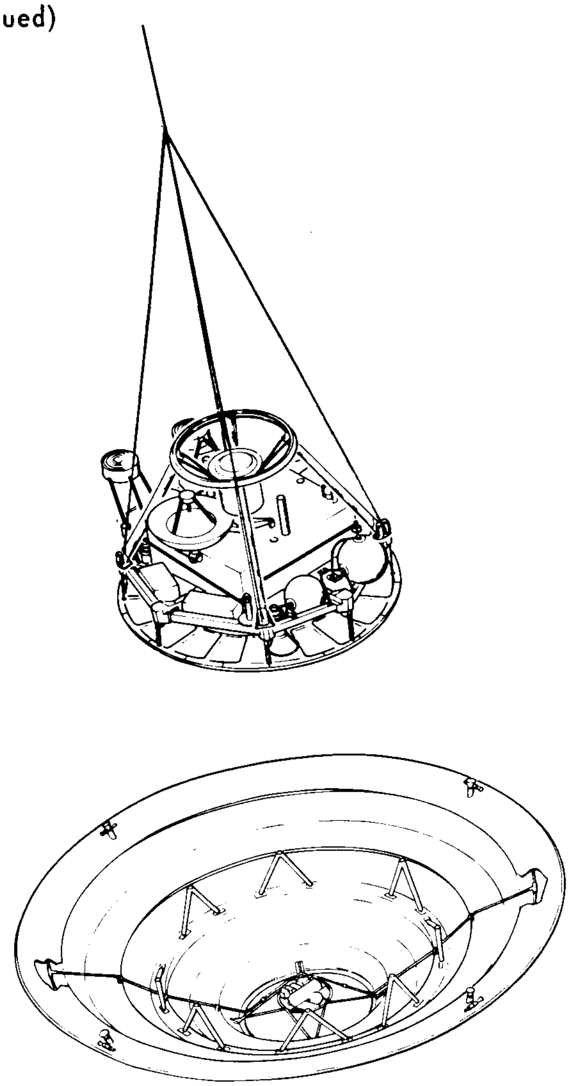


(6) DEORBIT MOTOR SEPARATION
- Fire Four Explosive Bolts to Release Spent Motor Case and Upper Support Structure
- Separated by Springs in Each Strut

CAPSULE BUS STAGING SEQUENCE
(Continued)

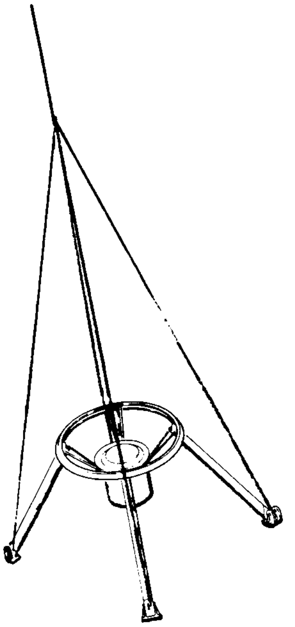


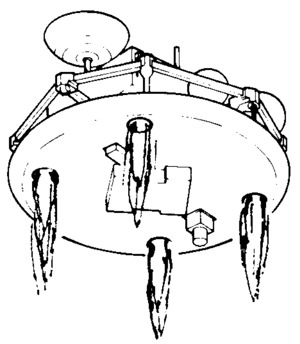
(7) PARACHUTE DEPLOYMENT
- Parachute Deployed by Catapult Firing Straight Aft § 100 FT/Sec Deploy Velocity
- Parachute Disreefed by Firing Four Pyrotechnic Actuated Reefing Cutters



(8) AEROSHELL SEPARATION
- Fire Four Explosive Bolts at Capsule Lander Aeroshell Interface
- Sequence at 8.0 Sec After Parachute Deployment

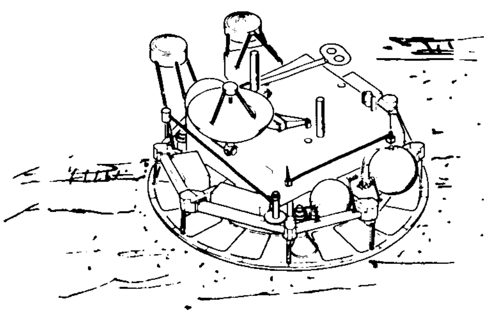
Figure 4.2-3 (Continued)





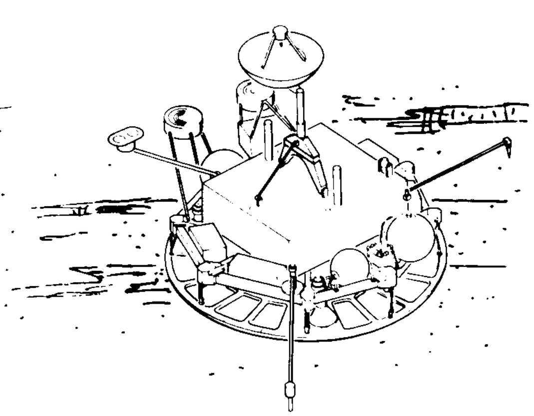
(9) PARACHUTE SEPARATION

- Ignite Terminal Propul System – Low Thrust
- Ignition Altitude is 5000 Feet
- After Successful Ignition, Fire Four Explosive Bolts at Base of Lower Support Structure



(10) LANDING

- Terminal Propulsion Terminated at 10 Feet
- Impact at $V_{v\,max}$ = 20 fps and $V_{h\,max}$ = 10 fps
- Surface Conditions Per Constraints Document
- Extend Stabilizing Blocks Released by Single Bolt Cutter Spring Actuated, Mechanically Locked



(11) LANDING OPERATION

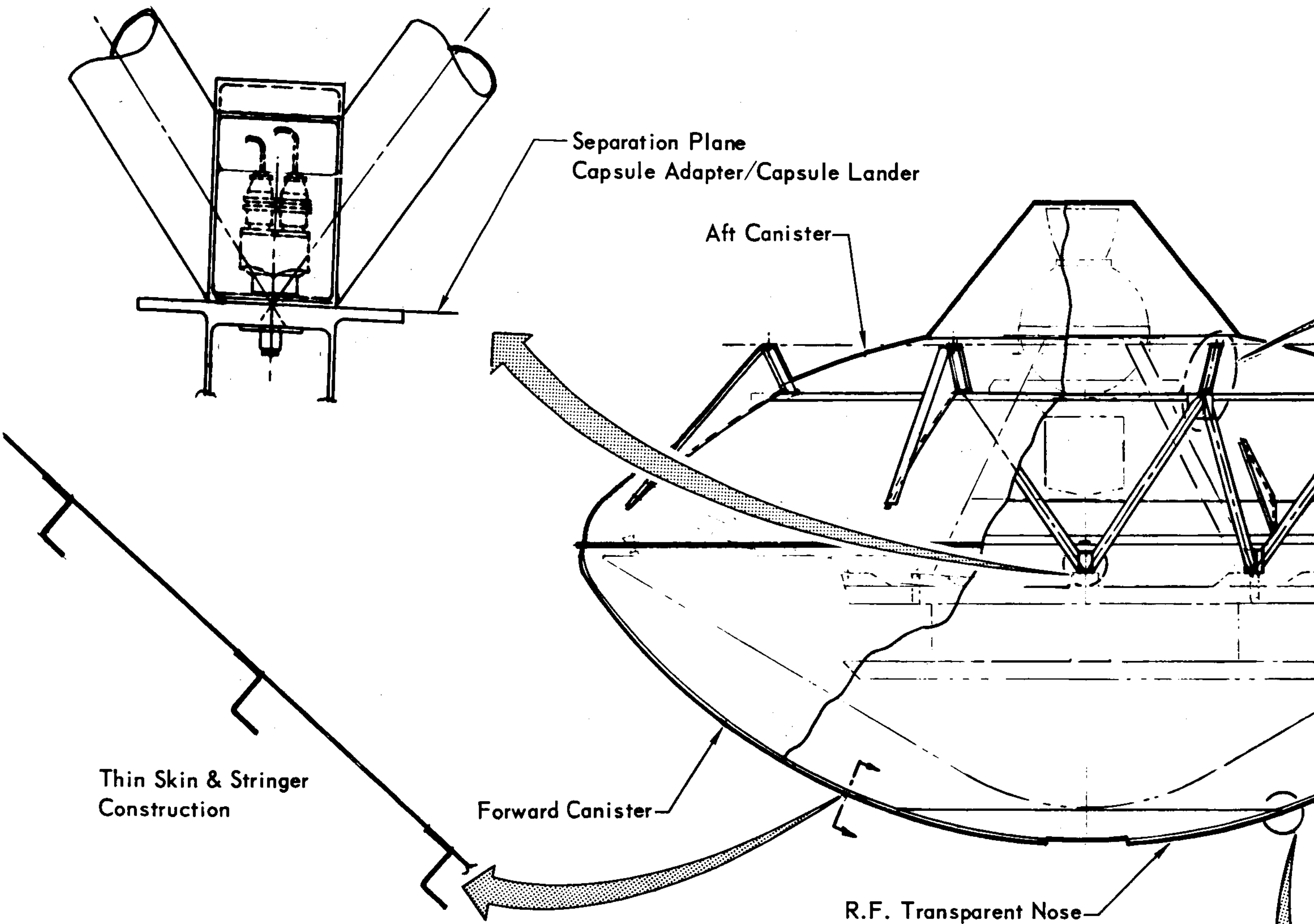
- Capsule Bus System Shut Down
- Surface Laboratory in Operation – All Equipment Deployed.

CAPSULE BUS
GROUP WEIGHT SUMMARY

	LAUNCH WEIGHT	DEORBIT PROPULSION INITIATION WEIGHT	ENTRY WEIGHT	TERMINAL PROPULSION INITIATION WEIGHT	TOUCH DOWN WEIGHT
Sterilization Canister & Adapter	543.0				
Deorbit Propulsion	523.3	523.3	18.3	18.3	18.3
Structure	(908.8)	(889.0)	(889.0)	(527.0)	(527.0)
Aeroshell	332.0	332.0	332.0	—	—
Lander (Includes Impact)	527.0	527.0	527.0	527.0	527.0
Separation Provisions	49.8	30.0	30.0	—	—
Ablative Heat Shield	204.4	204.4	204.4	—	—
Temperature Control	173.9	99.9	99.9	48.0	48.0
Attitude Control	56.2	55.2	51.6	—	—
Guidance & Control	132.7	132.7	132.7	121.9	121.9
Deployable Aero Decellerator	193.8	193.8	193.8	—	—
Terminal Propulsion	576.3	576.3	576.3	576.3	357.6
Telecommunications	146.3	98.7	98.7	98.7	98.7
Sequencer & Timer	50.8	26.9	26.9	26.9	26.9
Electrical Power	170.3	123.0	123.0	123.0	123.0
Capsule Bus Total Weight	3679.8	2923.2	2414.6	1540.1	1321.4

Figure 4.2-4

STRUCTURAL ARRANGEMENT CANISTER/ADAPTER

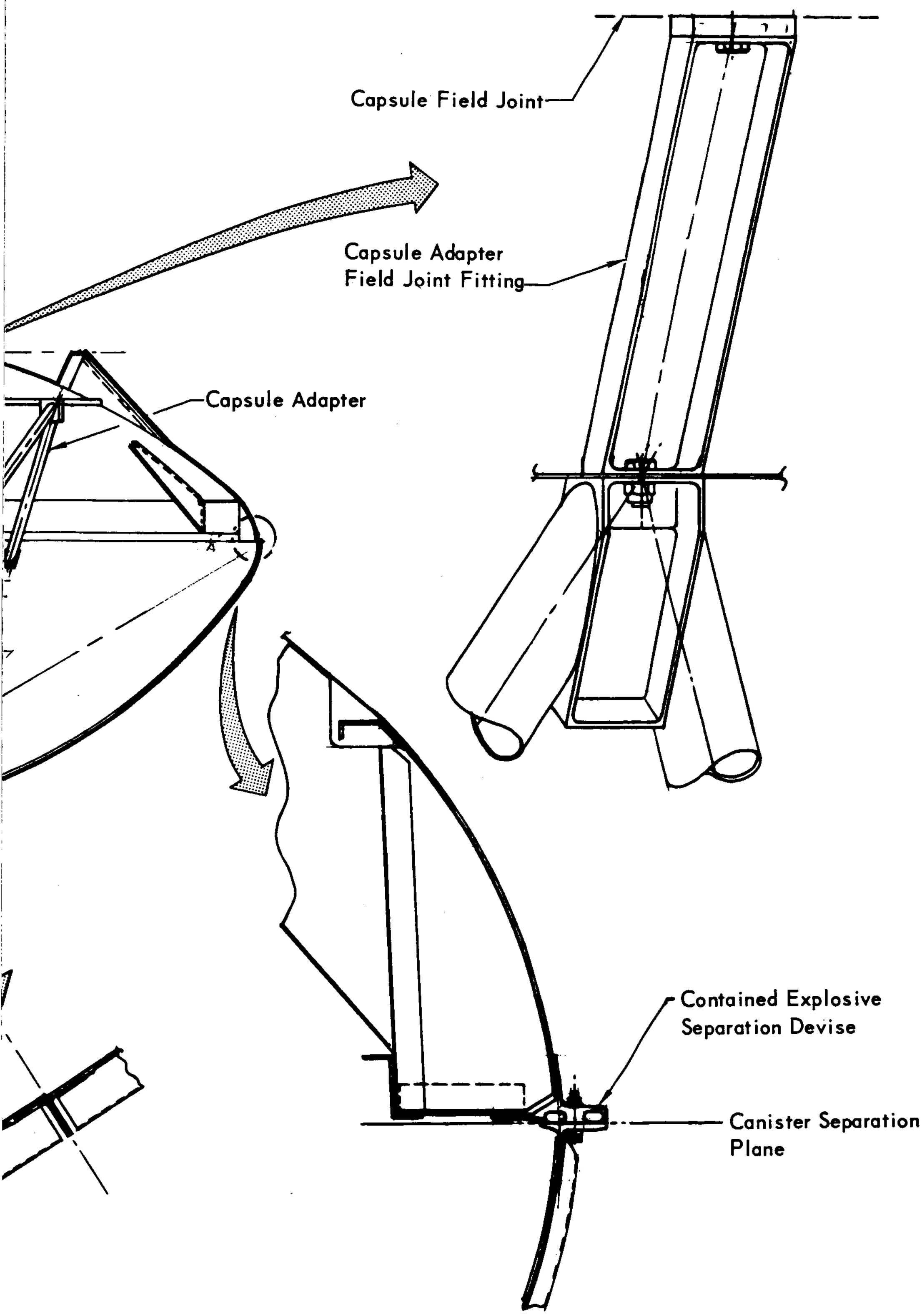


CANISTER TRADE STUDY SUMMARY

AREA	SELECTED APPROACH	ALTERNATES CONSIDERED	REASONS FOR SELECTION
Canister Pressure	2.25 psi Limit	1, 3, 5, 15 psi Limit Pressurization and Vent Systems	Near Minimum Structure Weight and Near Minimum Pressurization and Venting Complexity and Weight
Forward Canister Shape	Near Spherical	Conical Modified at Maximum Diameter by Radius	Spherical is Stiffer and Allows for More Capsule Growth (Volume)
Structure Material and Configuration	Aluminum (2024–TA) Semi-Monocoque	Beryllium, Titanium, Magnesium Steel, Fiberglass Materials; Monocoque, Honeycomb, Waffle, Ring Stiffened, Axial Corrugations Construction	Selected Approach is Lightest Weight Except Beryllium, Simple and Low Cost to Develop and Build.

FOLD-OUT #1

Figure 4.2-5


Capsule Field Joint
Capsule Adapter
Field Joint Fitting
Capsule Adapter
Contained Explosive
Separation Devise
Canister Separation
Plane

FOLD-OUT #2

from the adapter by use of the Capsule Bus reaction control system. The adapter and the aft canister remain with the Spacecraft.

4.2.1.2 Aeroshell/Heat Shield - The preferred Aeroshell design has been selected from many candidate approaches. Our selection was greatly influenced by the desire to use simple and familiar construction of standard materials, and to achieve maximum flexibility, light weight, and standardization for future missions.

As shown in Figure 4.2-6, the Aeroshell is a 120° sphere-cone with a 228.0 inch base diameter (preferred for standardization reasons). It has a single-face, corrugated titanium structure with closed triangular rings. The nose section is a combined radar altimeter antenna and ESP instrument head, which collects gas samples and measures pressure and temperature at the stagnation point. The instrument head is a flush, 9.50 inch diameter, beryllium block. A single pane, fused silica glass window is incorporated in the Aeroshell to allow visual imaging by the ESP during Capsule Bus entry.

We have selected a foamed reinforced methyl phenyl silicone ablator for the heat shield (except for the nose cap). This selection is based on an extensive comparative analysis - including feasibility tests - of various candidates, as summarized in Figure 4.2-7. Nose cap heat protection is nonablative, to prevent ablative products from interferring with the ESP imaging experiment and gas sampling. After evaluating various potential nonablative heat sinks, we have chosen for this purpose a phenolic fiberglass honeycomb sandwich, covered with hardened compacted fibers (Fiberfrax). (See Figure 4.2-8)

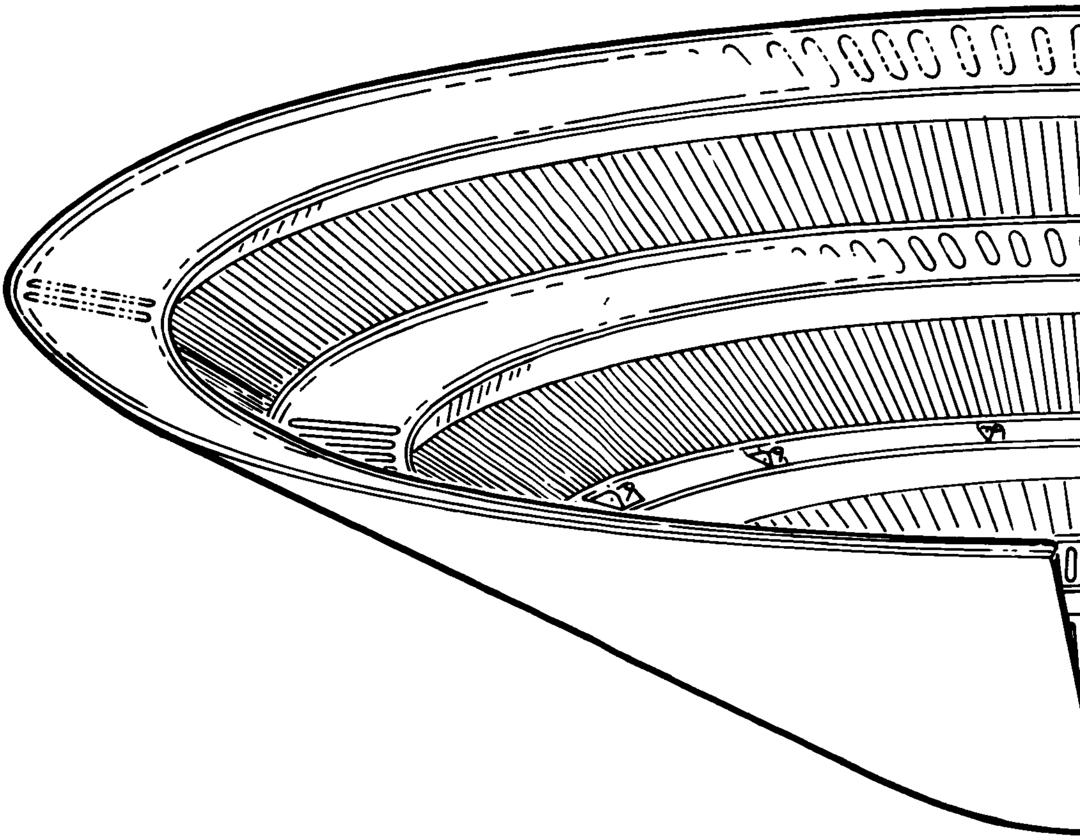
Control and maneuvering of the Capsule Bus from its separation from the Spacecraft to parachute deployment is accomplished through the reaction control subsystem, which is attached to the Aeroshell. Control is obtained from a pressurized monopropellant hydrazine reaction jet system, which has four 22 pound thrust chambers, 90° apart, for pitch and yaw control and four 2 pound thrust chambers for roll control.

The Aeroshell/heat shield is separated from the Capsule Lander by firing four explosive bolts at the interface. This is accomplished automatically by the Sequencer and Timer, 8.0 seconds after parachute deployment.

4.2.1.3 Capsule Lander - We consider the achievement of successful landing as the most important and most critical single function of the Capsule Bus. The problem here is to devise a landing system which can accommodate the widest range of surface conditions without failure or malfunction. We believe we have achieved this goal by our Uni-Disc Lander, which is a completely passive energy attenuator, specifically

PREFERRED STRUCTURAL CONCEPT

(19-FT



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4-22-1

AEROSHELL)

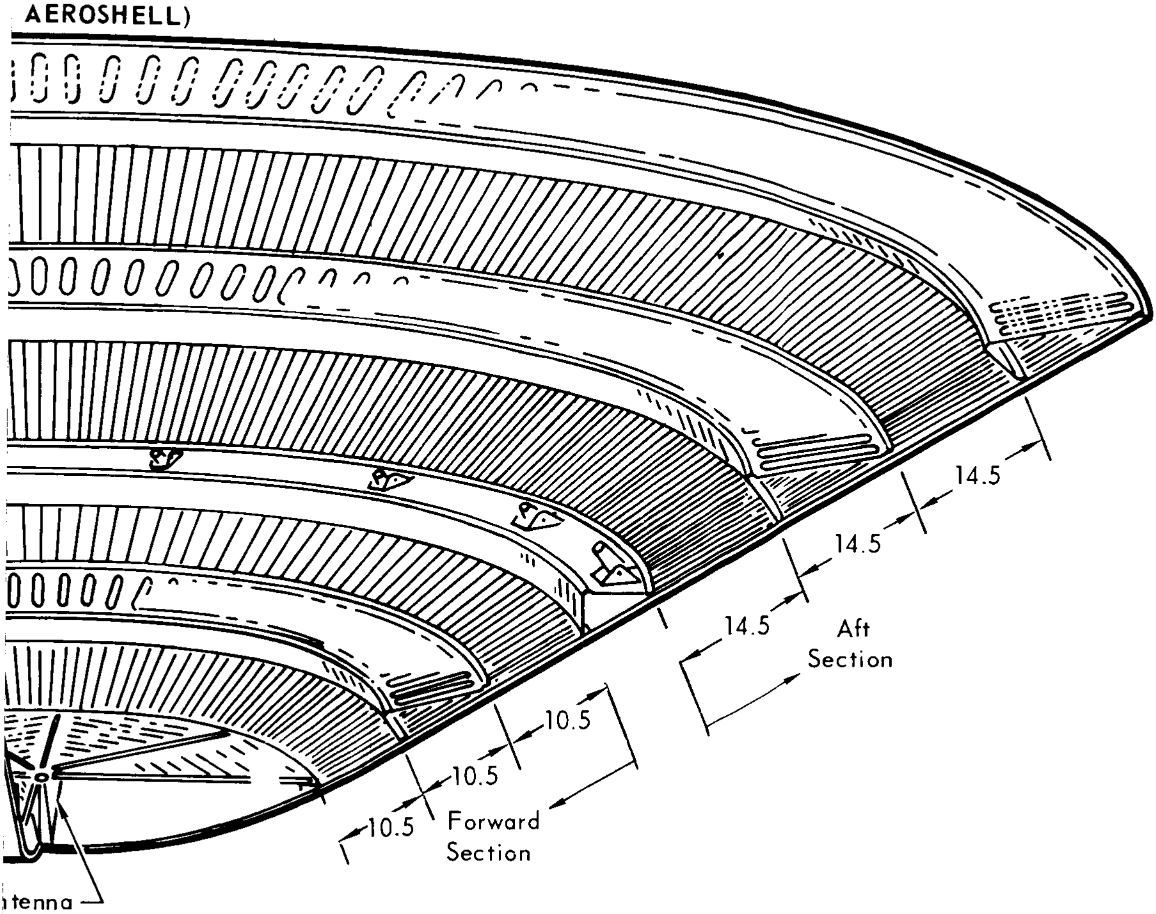


Figure 4.2–6

HEAT SHIELD ABLATIVE MATERIAL EVALUATION

PRIMARY CANDIDATES	DENSITY lb/ft³	EVALUATION FACTORS							
		RELATIVE THERMAL EFFICIENCY Heating Time for 500°F	STERILIZATION & HARD VACUUM EFFECTS	LOWER TEMP LIMIT	RELIABILITY CHAR RETENSION & BOND	TYPE OF BOND	FABRICATION TECHNIQUE	DEVELOP-MENT RISK	SELECTION
ESM 1004X Foamed Silicone Elastomer with Fibers	16.0	460 sec	Test Showed Negligible Effect	−300°F	Adequate for Mission	RTV 560 Paste Adhesive	Pre-formed as panels and applied	Moderate	Preferred
S-20T Silicone Elastomer Foamed into Prebonded Honeycomb	18.6	410 sec	Test Showed Negligible Effect	−300°F	Very Good	HT-424 Film Adhesive	Pre-Bond Honeycomb Core & Filled	Low	Preferred Development Backup
ESM 1030-1 Silicone Elastomer and Epoxy – Foamed	14.0	396 sec	Test Showed Negligible Effect	−70°F (Inadequate)	Poor Char Retention	RTV 560 Paste Adhesive	Pre-formed as panels and applied	High	
ESM 1030-2(S) Silicone Elastomer – Epoxy Foam in Split Honeycomb	18.4	260 sec	Test Showed Negligible Effect	−100°F (Inadequate)	Char Shrinkage and Spalling from Honeycomb	RTV 560 Paste Adhesive	Pre-formed as panels and applied	Very High	

Figure 4.2-7

SUMMARY OF COMPOSITE STRUCTURES FOR THE NOSE CAP

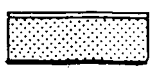
COMPOSITE CONFIGURATION		MATERIALS	COMMENTS
	Heatshield	Aluminum phosphate bonded fused quartz frabic. (For both heat-shield and structure)	Material properties have not been thoroughly investigated. Fabrication must also be evaluated. Will be considered further in Phase C.
	Structure		
	Heatshield	Flame-sprayed alumina and alumina foam.	Composite is too heavy. Reliability and temperature limit of the brittle external skin has not been established.
	Structure	Polybenzimidazole (PBI) – fiberglass laminate.	
	Heatshield	Hardened Fiberfrax	Recommended approach. Lightweight, uses state of the art fabrication techniques and permits the greatest versatility of fabrication.
	Structure	Phenolic-fiberglass honeycomb sandwich structure.	

Figure 4.2-8

designed to meet all the landing constraints and provides even further margin against tumbling. In arriving at the preferred concept, we have examined many approaches, as shown in Figure 4.2-9, from omnidirectional torus landers to conventional landing gear designs which have only limited anti-tumbling capability on even moderate surface slopes.

The Uni-Disc Lander is composed of three structural elements: the footpad, the shock attenuation ring, and the base platform. This is shown in Figure 4.2-10. The footpad is a 144 inch diameter disc. made of titanium skin and radial beams. Cutouts allow the terminal motors to fire through the lower surface. The radar altimeter antenna and the landing radar antenna are attached to the lower surface and are crushed on impact. The visual imaging camera for the Entry Science Package is also mounted on the lower surface and is pyrotechnically jettisoned prior to impact. Three stabilizing blocks are mounted flush in the footpad and are spring actuated and mechanically locked to further stabilize the lander after touchdown.

The impact attenuator is a cylinder of aluminum truss grid (honeycomb core) which is 2.1 inches thick and which crushes on impact. It is 13 in. high and 72 in. in diameter and is sandwiched between the footpad and base platform. Nominal crushing stroke is 5-6 inches for a 14 g landing. Eight pulley-ratchet assemblies, equally spaced around the periphery, connect the footpad and base platform to force total attenuator loading for a non-symmetrical landing.

The base platform is composed of eight titanium radial I-beams and serves as the mounting base for the Surface Laboratory, Entry Science Package, and the Capsule Bus support equipment.

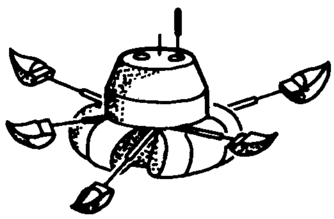
The four terminal propulsion engines (at a maximum thrust of 6800 pounds each) are mounted on the base platform, flush with the footpad. The nozzles are frangible and will be crushed during landing. Tankage is mounted on the base platform. The arrangement of equipment on the platform is such as to clear a rectangular Surface Laboratory that is symmetrically mounted and to allow an unobstructed view for the Surface Laboratory radiators.

The Capsule Lander supports the de-orbit motor and the stowed parachute by four struts that attach to the end of the base platform and straddle the Surface Laboratory. These are separated during descent by explosive bolts. The de-orbit motor itself and the upper section of the support structure are jettisoned by explosive bolts and springs, after motor cutoff and prior to entry.

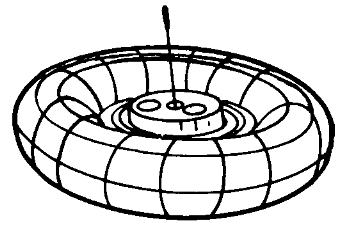
The entire base area is covered by a thermal curtain, made up of several layers of fiberglass, which protects the Capsule Bus equipment during entry.

VOYAGER CAPSULE LANDER CONCEPTS

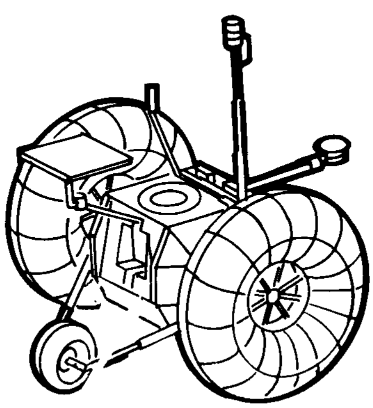
OMNIDIRECTIONAL



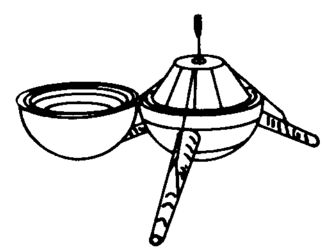
Mechanical Omnidirectional



Torus

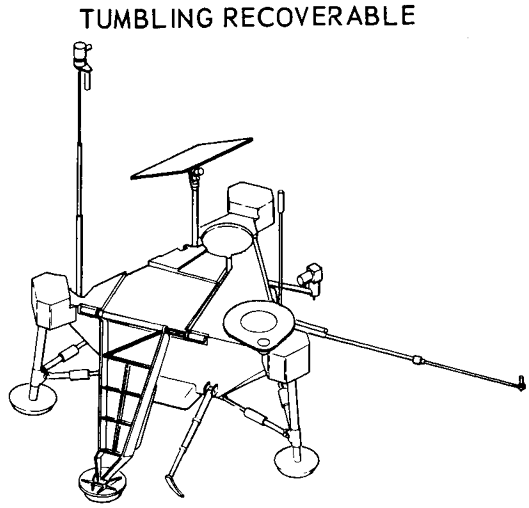


Torus Rover (Mobile



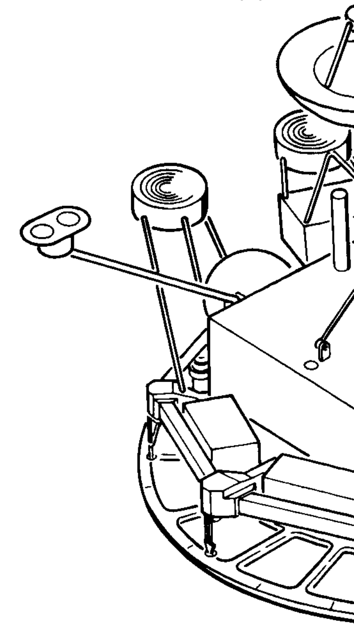
Spherical

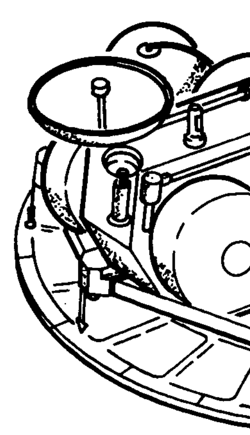
TUMBLING RECOVERABLE



Triangle

PREFERR

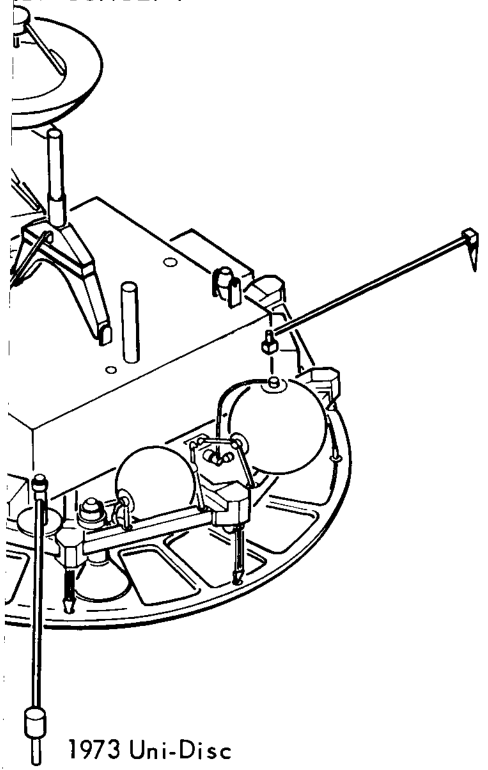




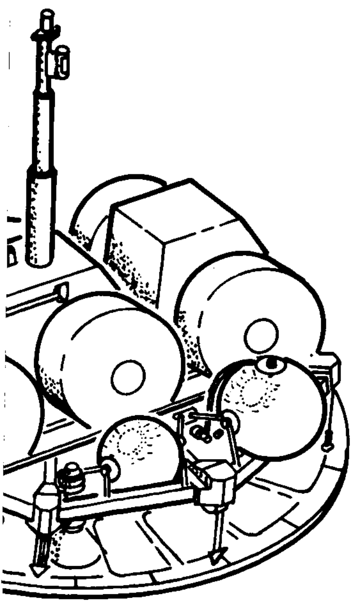
1979 Uni-Disc

Figure 4.2-9

RED CONCEPT

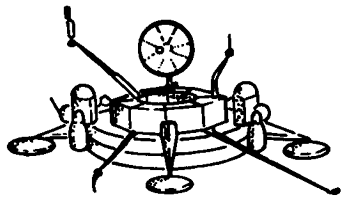


1973 Uni-Disc

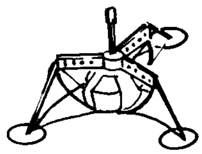


(Fully Mobile Laboratory)

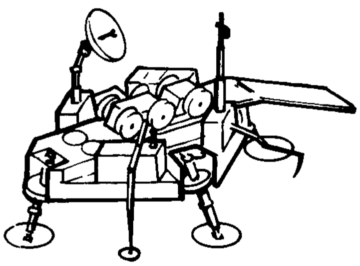
TUMBLING NON-RECOVERABLE



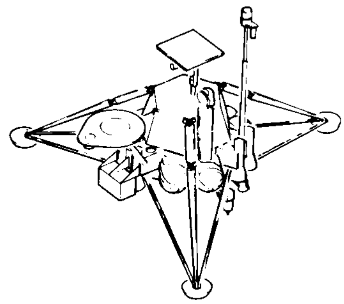
Stabilized Platform



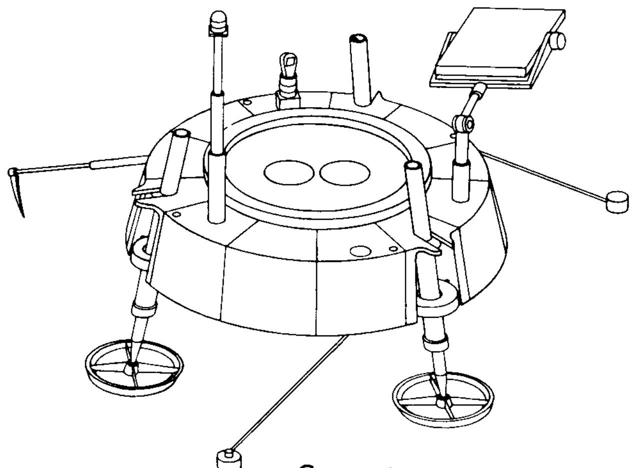
Pendulum



Modified Composite



Conventional



Composite

4-25-2

LANDER ATTENUATION ASSEMBLY

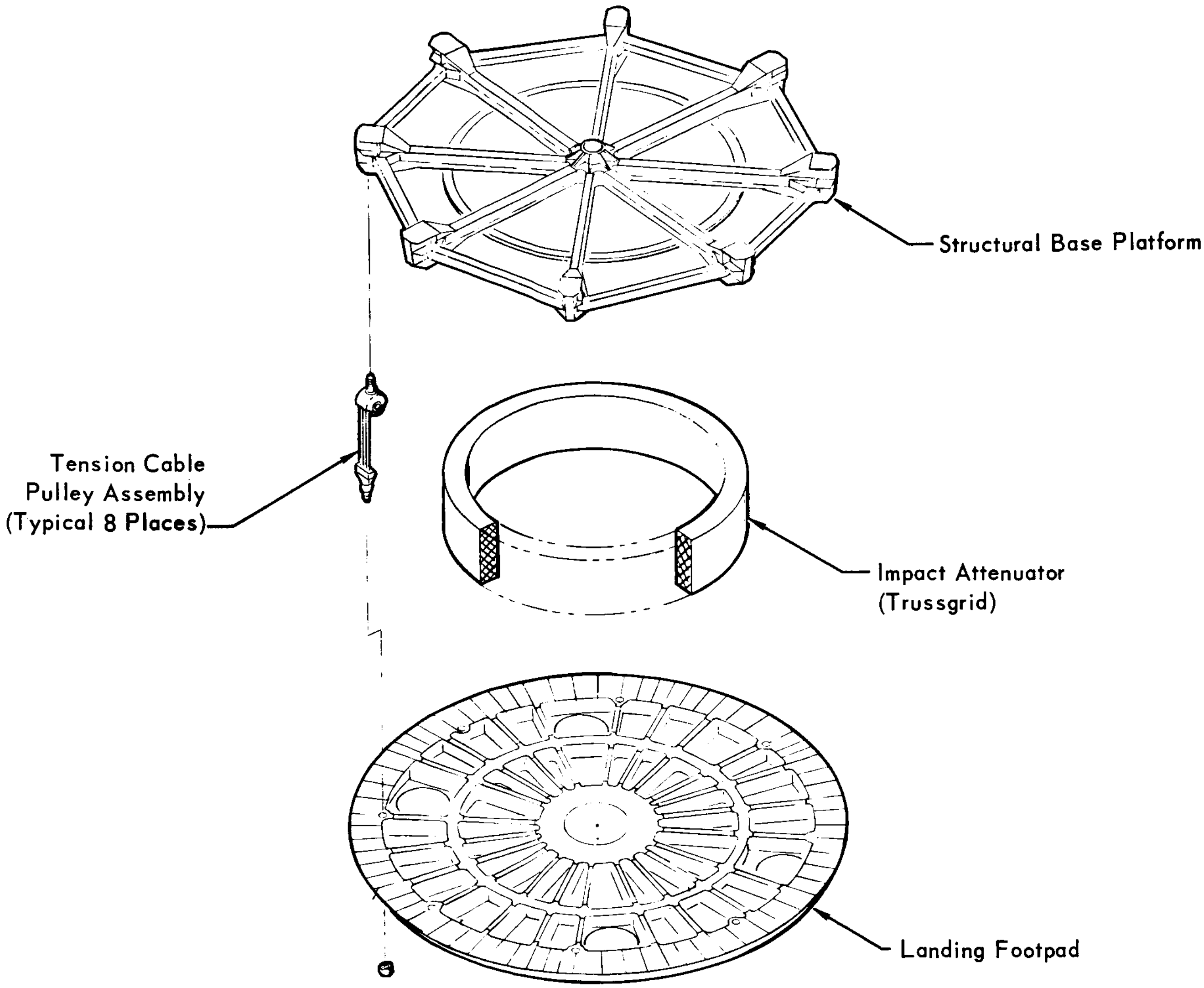


Figure 4.2-10

4.2.2 Major Supporting Subsystems

4.2.2.1 Guidance and Control - The guidance and control subsystem maintains an attitude reference, stabilizes the Capsule Bus at the desired attitude, turns the Capsule Bus to preprogrammed orientations as required, rolls the Capsule Bus at 3-4 revolutions per hour for thermal control during orbit descent, and provides the attitude and velocity reference for the lander terminal propulsion mode. It consists of an Inertial Measuring Unit (IMU) and a Guidance and Control Computer (GCC). The IMU is a rigid, machined block of aluminum which holds three floated rate-integrating gyros and a longitudinal (Z axis) accelerometer. The gyros use gas bearings that are insensitive to sterilization. The GCC is a general purpose machine which can accommodate a 4000-word program with an 8 microsecond add time. Parallel arithmetic is used throughout.

The guidance and control system is used in conjunction with the reaction control thrusters on the Aeroshell to provide attitude control during orbit and entry, and performs this same function during terminal descent by controlling the terminal propulsion subsystem. The operational aspects are shown in Figure 4.2-11 and a functional block diagram of our preferred concept is shown in Figure 4.2-12. The preferred approach of body-mounted sensors and a digital computer was arrived at after a comparative analysis of competing candidates, the results of which are summarized in Figure 4.2-13. Major advantages of our preferred strapped-down concept are:

a. Sterilizable hardware is currently in development.
b. Mission changes can be flexibly accommodated by computer software modifications.
c. This flexibility results in equipment standardization through 1979.

4.2.2.2 De-orbit Propulsion - The primary function of the de-orbit propulsion is to provide an impulse for deceleration (400-950 fps) of the Capsule Bus to place it on an entry trajectory into the Martian atmosphere. Our preferred design concept takes advantage of the simplicity and light weight of the solid propellant system; yet, through the feature of a jettisonable nozzle, has the thrust termination characteristics of a liquid propellant system (see Figures 4.2-14 and 4.2-15). The motor has been designed to permit center-of-gravity alignment, thus minimizing Capsule Bus response transients.

Standardization in the de-orbit propulsion has been accomplished by off-loading propellant in the 1973 version to allow for growth to the higher thrust requirements

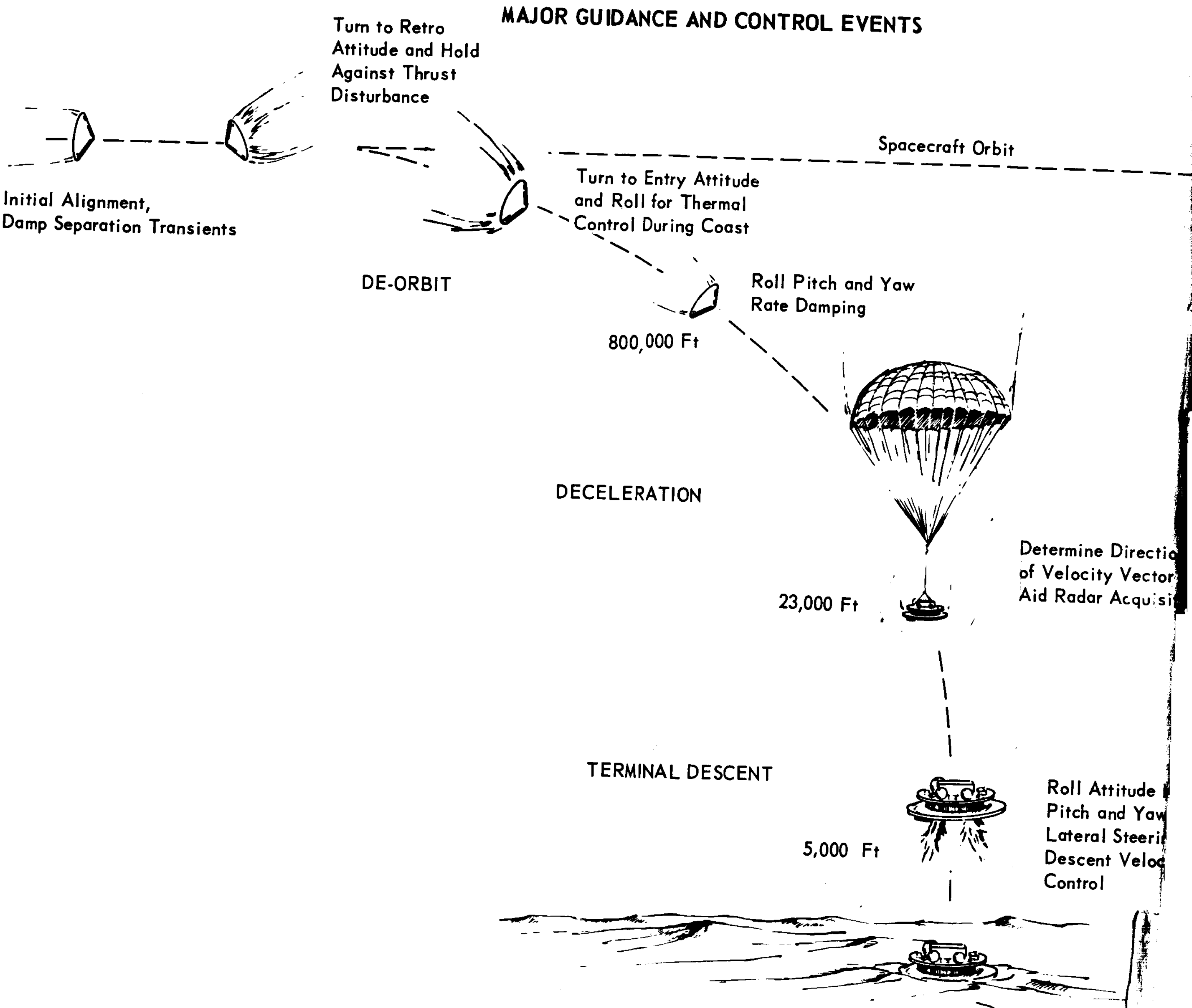

MAJOR GUIDANCE AND CONTROL EVENTS
Turn to Retro Attitude and Hold Against Thrust Disturbance
Initial Alignment, Damp Separation Transients
Spacecraft Orbit
Turn to Entry Attitude and Roll for Thermal Control During Coast
DE-ORBIT
Roll Pitch and Yaw Rate Damping
800,000 Ft
DECELERATION
Determine Directio of Velocity Vector Aid Radar Acquisi
23,000 Ft
TERMINAL DESCENT
Roll Attitude Pitch and Yaw Lateral Steeri Descent Veloc Control
5,000 Ft

Figure 4.2-11

FUNCTIONAL INTERFACES OF THE GUIDANCE AND CONTROL SUBSYSTEM

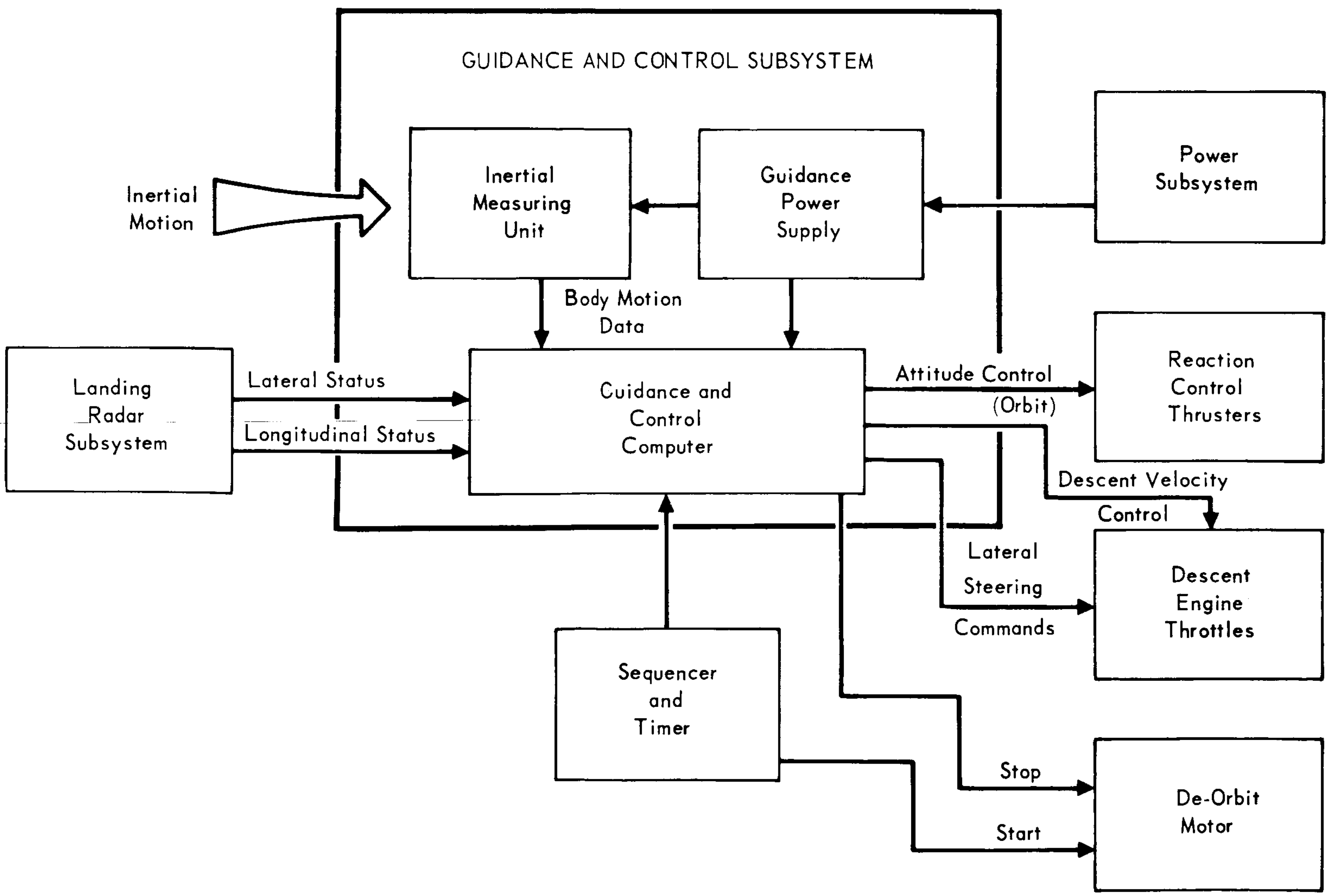


Figure 4.2–12

4-28-2

CAPSULE
GUIDANCE AND CONTROL SUBSYSTEM SUMMA

CANDIDATES	PROBABILITY OF MISSION SUCCESS	SYSTEM PERFORMANCE	DEVELOPM RISK
Strapdown IMU Digital Electronics	• High reliability • Key mission profile parameters may be changed in flight • Limited rate and attitude capability makes strapdown system more susceptible to unexpected wind and gust disturbances	• Pointing error is less than 0.87 degrees (3 sigma) • 1 degree per hour plus drift due to coning • 12 percent larger • 20 percent more power and weight than analog strapdown	• Digital str used on P • Digital au • Minimum s problems
Strapdown IMU Analog Electronics	• Simplest candidate, high reliability • Limited rate and attitude capability makes strapdown system more susceptible to unexpected wind and gust disturbances	• Pointing error is less than 1.74 degrees per axis at de-orbit (3 sigma) • Reference frame drift rate during limit cycle is greater than 1 degree per hour by coning drift error • Smallest size, least weight and power	• Strapdown Surveyor, • Sterilizabl developme • Potential analog ele
Gimballed IMU Analog Electronics	• Lowest reliability due to mechanical complexity and use of linear circuits. Not feasible to add significant redundancy to single IMU • High rate capability and wide attitude limits of gimballed platform make it least sensitive of all candidates to wind and gust disturbances	• Pointing error is less than 0.87 degrees (3 sigma) • 1 degree per hour (3 sigma) • 30 percent larger • 25 percent more power and weight than analog strapdown	• Two axis space prog • Platform b inertial se sterilizati
Gimballed IMU Digital Electronics	• Also has low reliability	• Pointing error is less than 0.87 degrees (3 sigma) • 1 degree per hour (3 sigma) • 25 percent larger • 20 percent more power • 25 percent more weight than analog strapdown	• Used on G • Similar co • Suitable p

Figure 4.2-13

BUS
ıRY COMPARISION OF ALTERNATIVES

/ALUATION CRITERIA			SELECTION
ENT	VERSATILITY	COST	
apdown guidance system RIME vehicle ʼopilot used on BGRV terilization induced	• Most flexible concept – both changes in mission profile and attitude freedom can be accommodated by changes to computer software	• Least uncertainty in cost estimate of this approach	X
gyros used on Ranger, Mariner and Lunar Orbiter e inertial components in nt sterilization problem with ctronics	• Analog electronics are inherently inflexible – changes in mission profile necessitate changes in circuitry	• Mission changes after development will significantly increase cost • Lowest initial cost	
platform used on classified ȷram earings, slip rings, and ʼnsors suitable for on must be developed	• Analog electronics are inherently inflexible	• Sterilizable platform development will be expensive	
·emini with analog autopilot ncept developed for BGRV latform must be developed	• High flexibility – software can accommodate changes in mission profile but range of attitude freedom is fixed	• Sterilizable platform development will be expensive • Highest initial cost	

DE-ORBIT PROPULSION

NOMINAL SUBSYSTEM WEIGHTS AND PERFORMANCE VALUES

ΔV = 950 FT/SEC
1979 WEIGHTS, OFF LOADED FOR 1973

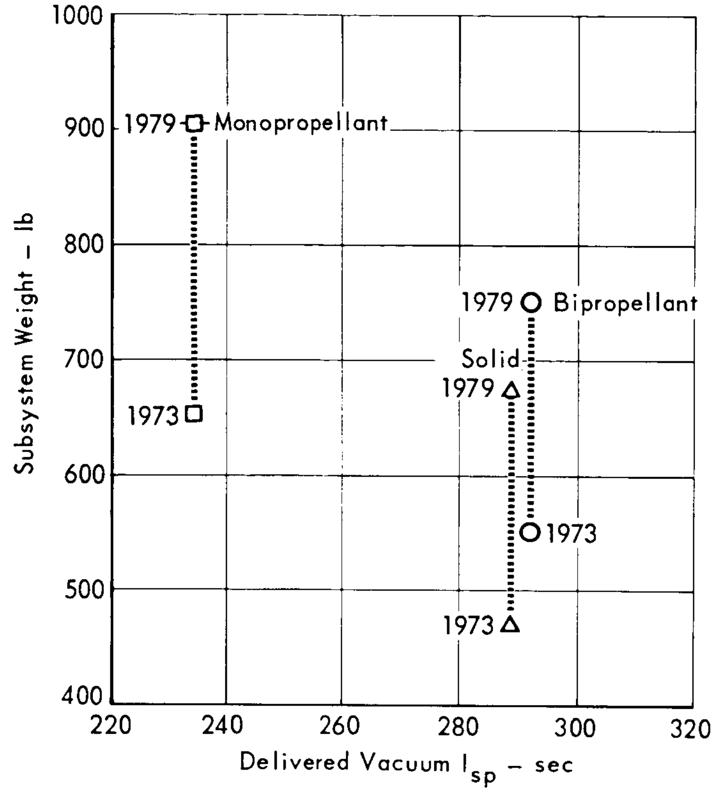


Figure 4.2–14

THRUST TERMINATION TECHNIQUE

NOZZLE EJECTION

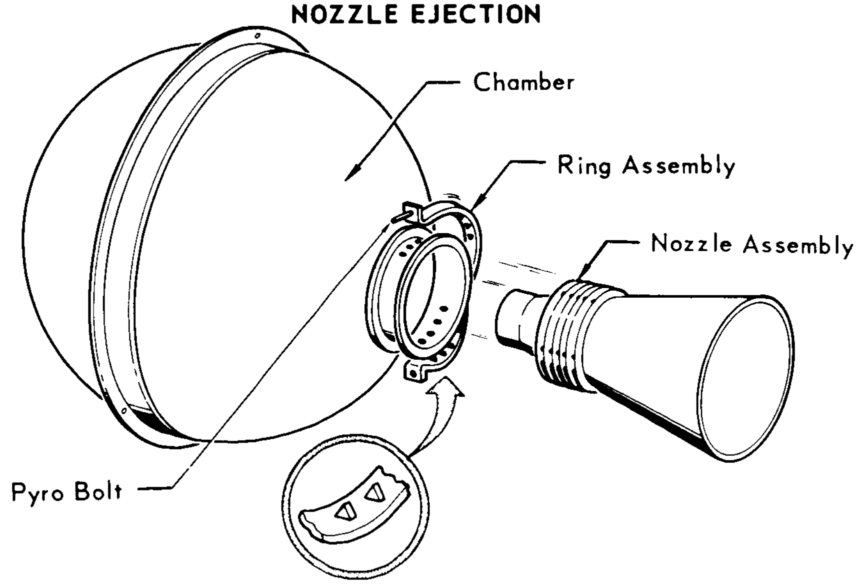


Figure 4.2–15

of the 1979 mission. Ejection of the de-orbit motor nozzle is accomplished by firing an explosive bolt which releases the thrust termination ring assembly.

Candidate competing concepts which were analyzed are shown in Figure 4.2-16. A functional block diagram of the preferred design is included in Figure 4.2-17.

4.2.2.3 Aerodynamic Decelerator - Our preferred Capsule Bus design incorporates an aerodynamic decelerator to provide the additional Flight Capsule deceleration required for safe initiation of the terminal descent maneuver. The desirability of using an aerodynamic decelerator, as opposed to an all-propulsive descent approach, is summarized in Figure 4.2-18.

Our design uses a parachute as the decelerator device. This parachute also separates the lander from the Aeroshell.

The key to the choice of a parachute lies in the expected Mach number of aerodynamic decelerator deployment and the development status of the PEP Program. For the 1973 operational envelope, deployment occurs below Mach 2. It is assumed that PEPP will demonstrate the feasibility of parachute deployment up to this Mach number. If future requirements dictate higher supersonic deployment Mach numbers - either due to newer Mars atmospheric data or other operational considerations - we believe that an attached tucked back Ballute offers an attractive alternative, with growth capability. Continuing effort in Ballute development is therefore recommended.

The performance characteristics of our preferred parachute system are shown in Figure 4.2-19 and its operational sequence is indicated in Figure 4.2-20.

4.2.2.4 Radar Subsystem - The radar subsystem is composed of a radar altimeter, which is used for high altitude measurements, and a landing radar, which is used for terminal control. The radar altimeter is turned on just prior to entry and provides continuous altitude measurements from 200,000 feet down for science data correlation, and altitude marks for turning on various subsystems and release of equipment (parachute, Aeroshell, etc.). Terminal control is provided by the landing radar (in conjunction with the guidance and control subsystem) by slant range and velocity measurements from 5000 feet down to 10 feet. The altimeter provides back-up for the landing radar range measurement from 5000 to 50 feet.

The landing sequence is shown in Figure 4.2-21.

The radar altimeter has two antennas: one mounted on the Aeroshell and the other on the lander for use after Aeroshell separation. It operates at L-Band and uses noncoherent pulse modulation with a peak transmitter power of 500 watts. It has a hemispherical antenna pattern, to accommodate variations in flight path angle without requiring roll control.

CAPSULE BUS
DE-ORBIT PROPULSION SUBSYSTEM SUMMARY COMPARISON OF ALTERNATIVES

CANDIDATES	EVALUATION CRITERIA					SELECTION
	PROBABILITY OF MISSION SUCCESS	SYSTEM PERFORMANCE	DEVELOPMENT RISK	VERSATILITY	COST	
Solid Propellant	• Highest Reliability = .9950 • Armed prior to launch • Simple Design	• Lowest Weight = 460 Lbs. (477 Lbs for 1979 Off-Loaded Design) • Low Volume = 10.2 Ft^3 • High I_{sp} = 287 sec • High Mass Fraction = .885	• High – Requires propellant development and full scale demonstration	• Very versatile with thrust termination. • The 1979 design can be off-loaded with minimum penalty.	• Lowest	X
Monopropellant	• High Reliability = .9C40 • Requires propellant ullage orientation maneuver	• Highest Weight = 632 Lbs. • High Volume = 12.0 Ft^3 • Low I_{sp} = 235 Sec • Moderate Mass Fraction = .830	• Moderate – N_2H_4 and storage tank compatibility requires development for 275°F. Catalyst bed and ETO compatibility must be resolved.	• Very versatile. Termination easily effected. • Off-loading for '73 results in low penalty.	• Moderate	
Bipropellant	• Low Reliability = .9918 • Requires propellant ullage orientation maneuver	• Moderate Weight = 533 Lbs. • Low Volume = 9.8 Ft^3 • High I_{sp} = 292 Sec • Low Mass Fraction = .798	• Low – More propellant/mat'ls compatibility testing req'd but NTO/MMH propellants are thermally stable @ 275°F.	• Very versatile. Termination easily effected. • Off-loading for '73 results in low penalty	• Highest	
Composite – Common Tanks	• Low Reliability = .9926 • Requires positive expulsion tankage • Interface Complexity – engine jettison – sever propellant lines.	• High Weight = 607 Lbs. • Requires Large Volume Tanks • High I_{sp} = 292 Sec • Low Mass Fraction = .700	• Same as Bipropellant	• Versatile. Termination easily effected. • However separate tanks must be designed for the '73 and '79. Positive expulsion req'd.	• High	
Composite – Common Tanks & Engines	• Low Reliability = .9869 • Requires ullage maneuver • Interface Complexity – Use of terminal engines requires firing through Aeroshell	• High Weight = 598 Lbs. • Requires Large Volume Tanks • High I_{sp} = 295 Sec • Low Mass Fraction = .698	• Same as Bipropellant	• Same as Composite above except for positive expulsion devices which are not required.	• Moderate	

Figure 4.2-16

DE-ORBIT PROPULSION SUBSYSTEM
FUNCTIONAL BLOCK DIAGRAM

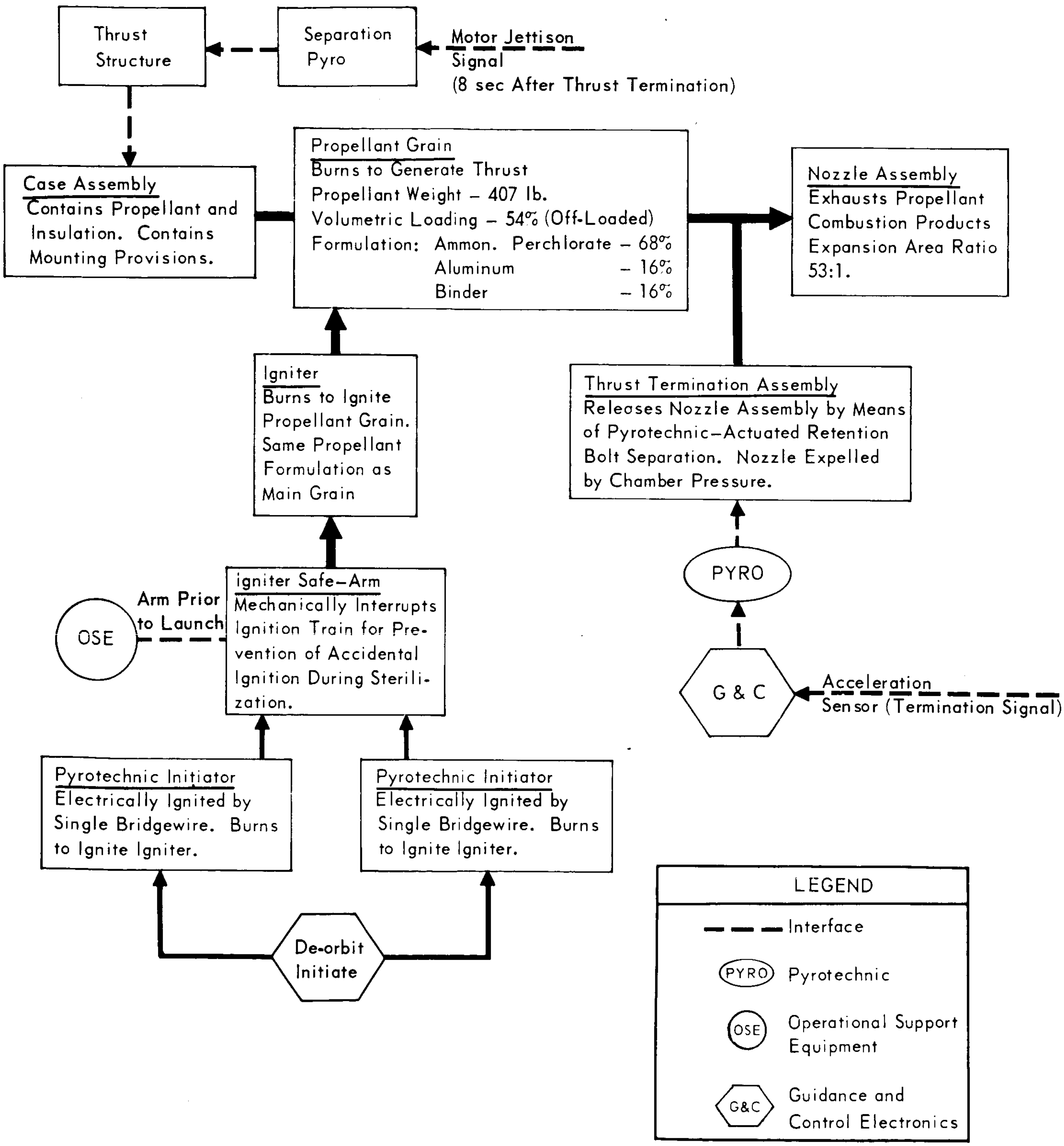


Figure 4.2–17

CAPSULE BUS

DECELERATOR CONFIGURATION SUMMARY COMPARISON OF ALTERNATIVES

CANDIDATES			
TYPE	CHARACTERISTICS	PROABILITY OF MISSION SUCCESS	SYSTEM PERFORMANCE
Propulsion & Aerodynamic Decelerator	Supersonic Solid Parachute – 4 Engine Terminal Propulsion Subsystem – Differential Drag Separation	• Short burn = 50 sec • Reliability = .9699 (Parachute reliability = .996) • Maximum wind drift velocity • Standard sequence	• Total decelerator weight = 760 lb • Volume of TPS = 7.85 ft^3 • Volume of parachute = 2.54 ft^3
All Propulsion	4–Engine Terminal Propulsion Subsystem Fire-Through Holes	• Longest burn = 70 sec • Reliability = .9653 • Torquing maneuver required at separation • Recontact potential • At landing radar limit	• Terminal propulsion subsystem weight = 890 lb • Volume terminal propulsion subsystem = 14.26

Figure 4.2–18

EVALUATION CRITERIA			
DEVELOPMENT RISK	VERSATILITY	COST	SELECTION
• High risk with two parallel development • Can fall-back to all-propulsive configuration • 3–1/2 yr for terminal propulsion subsystem	• High level of versatility afforded by combination decelerators • $M \leq 2$ deployment • Ignition altitude = 5k ft • Simplified Aeroshell separation	• Parachutes = \$17.1 million • Engines = \$80.5 million	X
• High risk (lower than combination development • 3–1/2 yr for terminal propulsion subsystem	• Limited because of large operating window required • Ignition altitude = 15k ft • Aeroshell separation complex	• Bipropellant engines = \$81.5 million	

CAPSULE LANDER
PARACHUTE PERFORMANCE CHARACTERISTICS

PARAMETER	PERFORMANCE CHARACTERISTICS	
	VALUE (MAXIMUM / MINIMUM)	CRITICAL MODEL ATMOSPHERE
Deployment Altitude	23,000 Ft.	
Deployment Mach No.	2.0 / .43	VM – 8 / VM – 9
Deployment Dynamic Pressure	13.2 psf / 3.65 psf	VM – 8 / VM – 9
Catapult Velocity	100 Ft/Sec	
Opening Shock Load (Reefed)	18,300 Lb. / 6200 Lb.	VM – 8 / VM – 10
Shock Load (Full Open)	18,300 Lb. / 9200 Lb.	VM – 8 / VM – 10
Time From Parachute Deployment to Aeroshell/Lander Separation	12.0 Sec.	
Altitude at Aeroshell/Lander Separation	18,900 Ft. / 15,600 Ft.	VM – 10 / VM – 7
Altitude at Parachute Release	5000 Ft.	
Lander Velocity at 5000 Ft. Terminal Propulsion Initiation	283 Ft/Sec / 116 Ft/Sec	VM – 7 / VM – 10
Lander Altitude When Aeroshell Impacts Martian Surface	6700 Ft. / 4170 Ft.	VM – 10 / VM – 7
Lander Surface Impact Velocity on Parachute (Terminal Propulsion Failure)	271 Ft/Sec / 112 Ft/Sec	VM – 7 / VM – 10

NOTE: The entry conditions for critical value are V_e = 13,000 Ft/Sec & $\gamma_e = -20^\circ$

Figure 4.2–19

AERODYNAMIC DECELERATOR SUBSYSTEM OPERATIONAL SEQUENCE

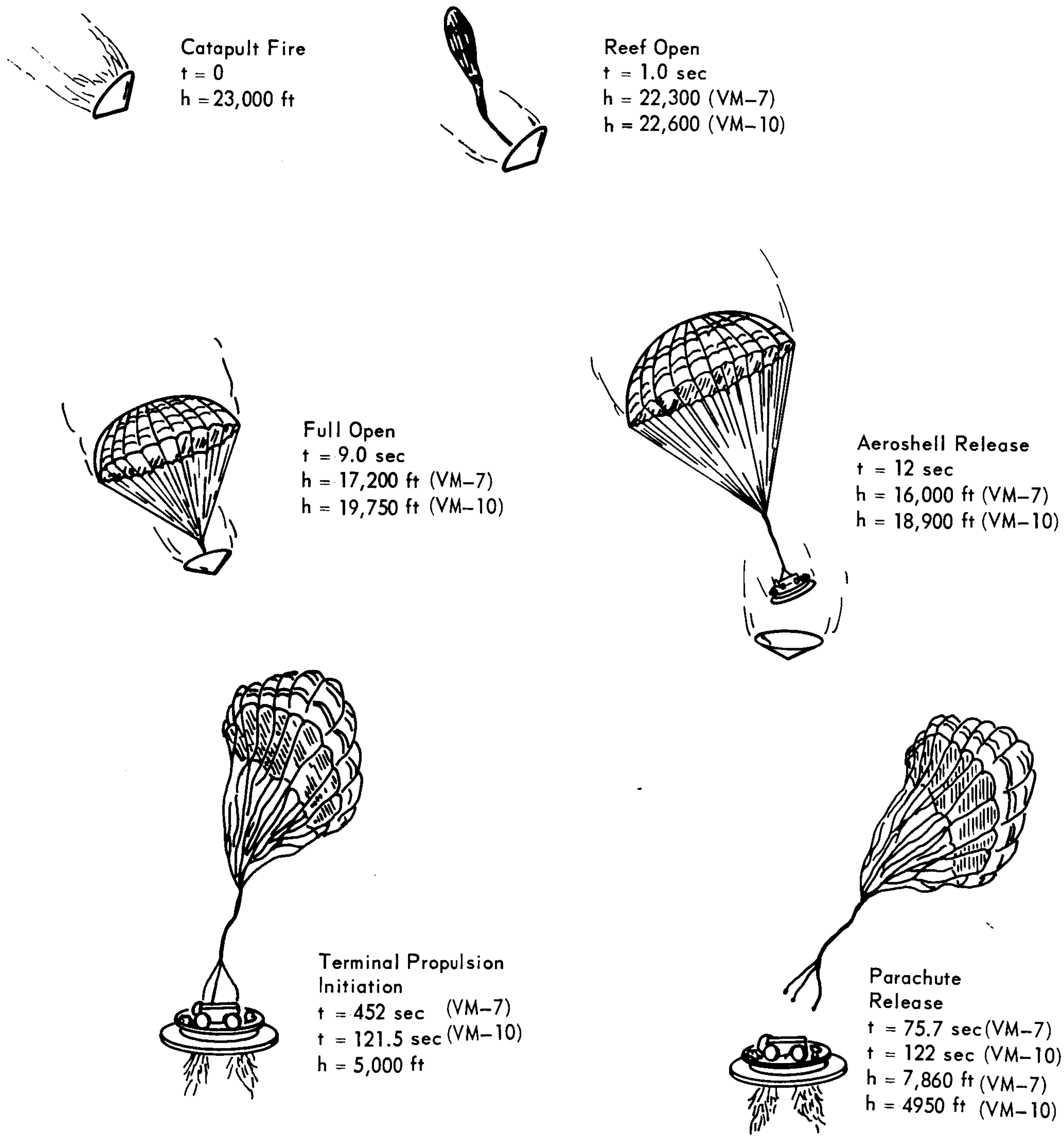


Figure 4.2–20

RADAR SUBSYSTEM OPERATION

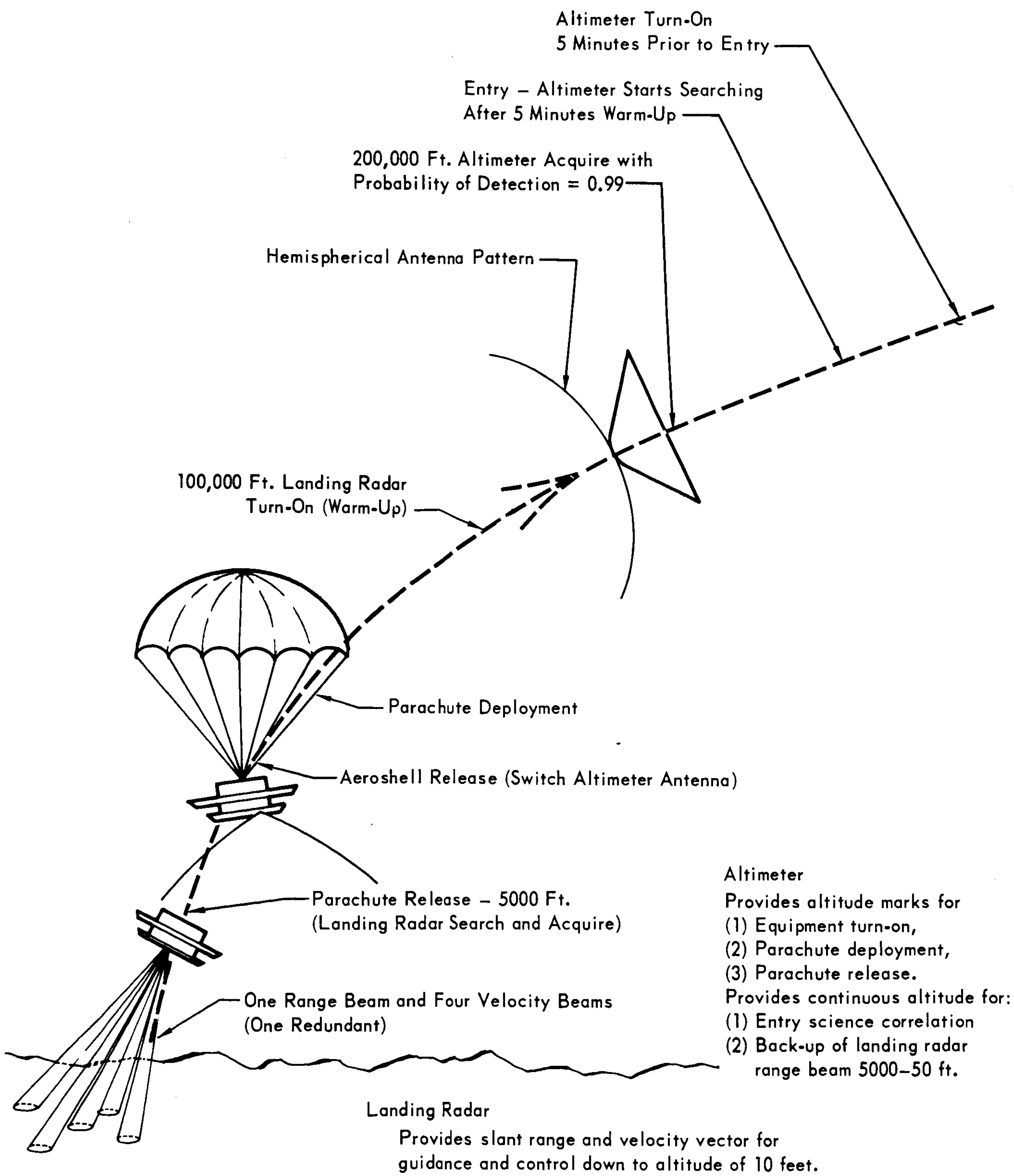


Figure 4.2–21

The landing radar employs one 5° wide range beam (directed along the roll axis to eliminate sensitivity to roll position) and four 5° wide velocity beams, placed 20° from the roll axis, this being patterned after the proven LEM radar concept. The velocity beams are placed symmetrically around the roll axis. Only three out of the four velocity beams are required for velocity vector measurement. The landing radar operates at X-Band, with separate frequencies for measuring range and velocity. The range channel uses linear FM-CW modulation, but the velocity channels operate with an unmodulated carrier. All five receiver channels utilize similar frequency trackers. A data converter processes the frequency tracker outputs to extract the capsule orthogonal velocity components (Vx, Vy, and Vz), and removes the velocity component from the range tracker output. At 2500 feet, the frequency, deviation on the FM-CW range beam is increased and the bandwidth of the range beam tracking filter is decreased. This provides increased range accuracy. At a prescribed velocity (derived from a combination of Doppler frequency and slant range), the filter bandwidth of the velocity beam tracking filter is also decreased, resulting in improved accuracy of the velocity measurement.

Alternative approaches to the preferred radar subsystem are summarized in Figure 4.2-22 and a functional operational diagram is shown in Figure 4.2-23.

4.2.2.5 Terminal Propulsion - Terminal propulsion of the Capsule Lander is initiated just prior to parachute release. This is preferred because the parachute provides a certain amount of descent mode capability in case of a failure in propulsion ignition. The initiation signal emanates from the radar altimeter.

The terminal propulsion subsystem is a storable hypergolic bipropellant liquid rocket system, with the propellant supplied to the thrust chamber by pressurized helium gas. The major components are shown in Figure 4.2-24 and consist of four fixed engine assemblies and their supporting equipment. The subsystem provides a controlled terminal descent, including pitch and roll control, from 5000 feet to 10 feet above the surface of Mars.

Extensive trade-off studies were conducted to arrive at this preferred concept. A summary comparison chart of the four high value candidates remaining from an initial choice of 18 alternatives is shown in Figure 4.2-25. Our preferred choice was guided predominantly by:

a. A weight saving and reduction in complexity, resulting from the elimination of a separate gimballing system or reaction control system.
b. Greater compatibility with the Capsule Lander design configuration.
c. Smaller exhaust plumes, reducing interference with the landing radar pattern.

CAPSULE BUS
RADAR SUBSYSTEM OPTIMIZATION STUDY SUMMARY

	FUNCTION (CHARACTERISTIC)	CANDIDATES	EVALUATION FACTORS: COMPLEXITY	PERFORMANCE	SIZE & WEIGHT	POWER (EFFICIENCY)	DEVELOPMENT STATUS	DEPENDENCE ON OTHER FUNCTIONS	SELECTION & REMARKS
Altitude Measurement	• Antenna Configuration	Steered Pencil Beam	Most Complex	Limited Angular Coverage	Comparable	Least Trans Power	Comparable	Sense Roll Position	
		Switched Fan Beam	Nominal Complexity	Acceptable		Nominal Trans Power		Sense Roll Position	
		Single Element Radiator	**Least Complex**	**Acceptable**		**Most Trans Power**		**None**	**Simplest with No Roll Control**
	• Operating Frequency	20 MHz to L-Band		No Breakdown	Larger (Antennas)	Least Trans Power			
		L-Band	**Comparable**	**Breakdown Manageable**	**Minimum Total**	**Nominal Trans Power**	**Comparable**	**Same**	**Minimum Total Size & Weight**
		Above L-Band		Breakdown Problems	Larger (Batteries)	Most Trans Power			
	• Modulation Technique	FM/CW, ICW, and PN-Coded CW	More Complex	Trans/Receiver Leakage		Less Efficient	Adequate Development	Not Applicable	
		Non-coherent Pulse	**Less Complex**	**More Accurate (Leading Edge Tracking)**	**Comparable**	**More Efficient**	**Most Developed**		**Accuracy, Simplicity, and Development Status**
Velocity Vector Measurement	• Beam Configuration	3 Fixed Beams	Nominal Complexity	Have Included Redundancy for Comparable Reliability	Heavier		Most Developed	More Sensitive to Pitch Without Roll Control	
		4 Fixed Beams (One Redundant)	**Nominal Complexity**		**Lightest For Same Reliability**	**Comparable**	**Extension of Above Three Beam System**	**Less Sensitive to Pitch Without Roll Control**	**Reliability vs. Weight and Insensitivity to Pitch Angle**
		Scanned or Switched Beam	Most Complex		Heaviest		Least Developed	Most Sensitive to Pitch Without Roll Control	
	• Beam Position	**4 Beams Spaced Symmetrically Around Roll Axis**	**Simpler Processing Circuitry to Derive V_x, V_y, V_z**	**Comparable Without Roll Control and with One Beam Out**	**Comparable**	**Comparable**	**Comparable**	**Pitch Angle Sensitivity Equal for Both Configurations with Worse Roll and Beam Out**	**Simpler Processing Circuitry**
		3 Beams Spaced Symmetrically Around Roll Axis, 1 Beam Along Roll Axis	More Complex Beam Processing Circuitry					Less Sensitive to Pitch Without Roll Control, Providing that All Channels are Working	
	• Beam Cone Angle	Much Less than 20 deg		Poor Lateral Velocity Accuracy				Least Sensitive to Pitch Angle	
		Approximately 20 deg	**Same**	**Nominal Lateral Velocity Accuracy**	**Comparable**	**Comparable**	**Comparable**	**Nominal Pitch Angle Sensitivity**	**Acceptable Lateral Velocity Accuracy and Pitch Sensitivity**
		Much Greater than 20 deg		Better Lateral Velocity Accuracy				Most Sensitive to Pitch Angle	
	• Modulation Technique	**Unmodulated CW**	**Least Complex**	**Adequate Performance with Spurious Signals**			**Most Developed**	**Velocity Measurement Independent of Range Channel**	**Simplest, Performance Adequate with Preferred Landing Concept**
		Modulated by Range Channel	Nominal Complexity	Better Spurious Signal Rejection – Zero Range	Comparable	Comparable	Less Developed	Velocity Measurement Depends on Range Channel	
		Modulated Independently	Most Complex	Better Spurious Signal Rejection – Zero Range			Less Developed	Velocity Measurement Independent of Range Channel	
Slant Range Measurement	(Beam Along Roll Axis)	ICW	Less Complex			More Efficient Use of Trans Power	Less Developed	Range Measurement Independent of Velocity Measurement	
	• Modulation Technique	Sinusoidal FM/CW (Sideband Processing)	Most Complex	Comparable	Comparable	Least Efficient Use of Trans Power	Less Developed	Range Measurement Independent of Velocity Measurement	
		Linear FM/CW (Sawtooth)	**Less Complex**			**More Efficient Use of Trans Power**	**Most Developed**	**Range Measurement Depends on Velocity Measurement**	**Simplicity, Efficiency, and Development Status**

Figure 4.2-22

RADAR SUBSYSTEM INTERFACE DIAGRAM CAPSULE BUS

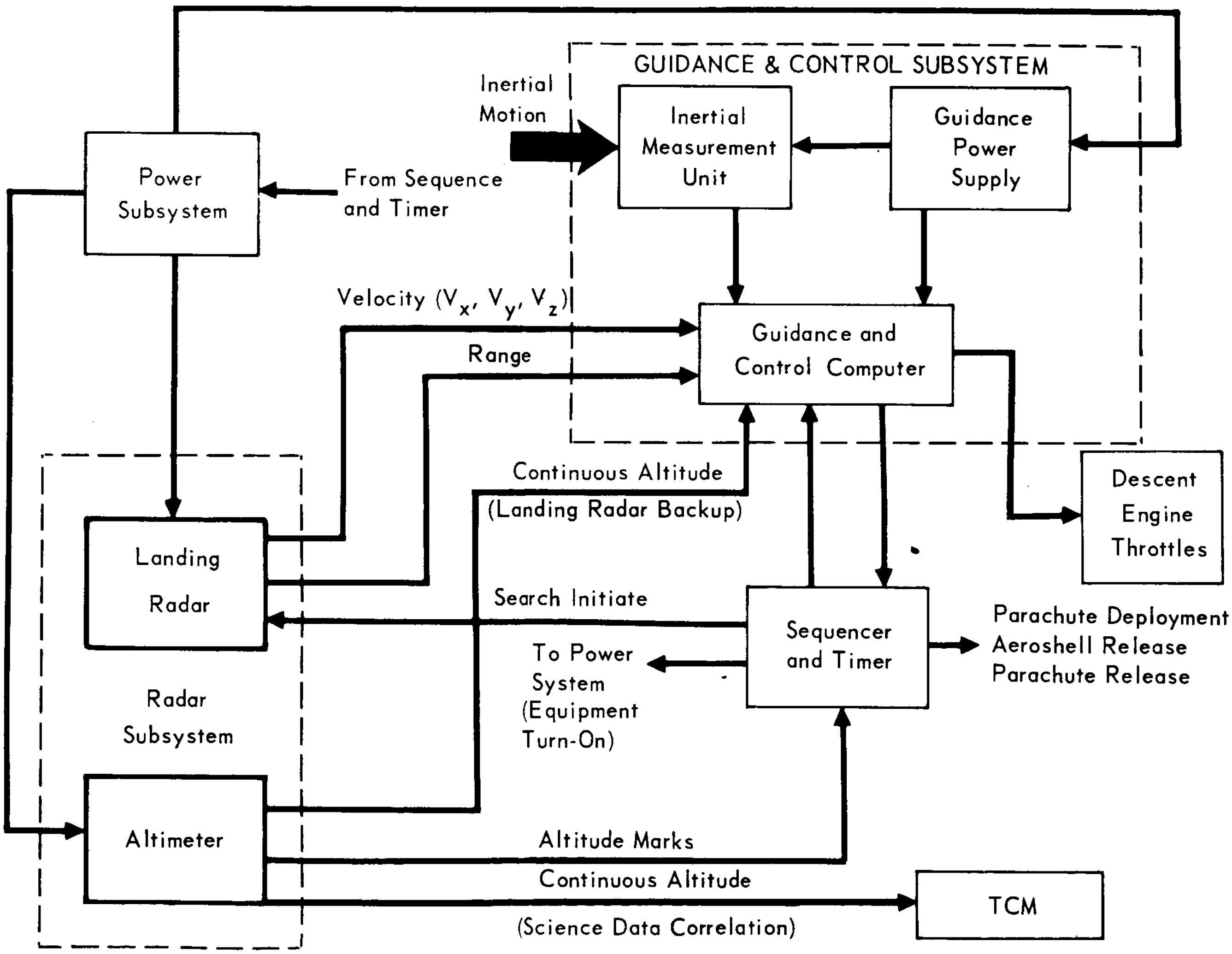


Figure 4.2-23

TERMINAL DESCENT PROPULSION SUBSYSTEM MAJOR COMPONENTS

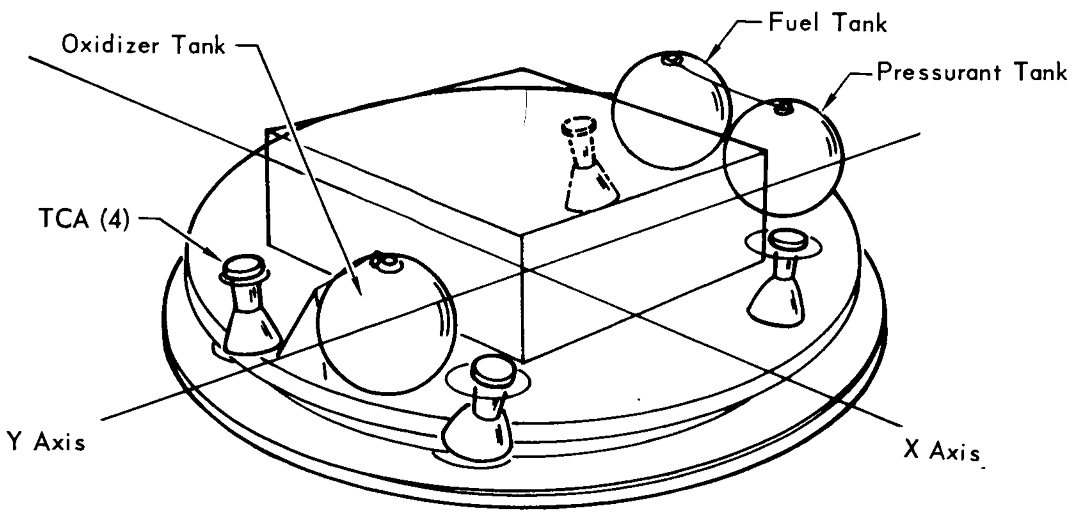


Figure 4.2-24

CAPSULE BUS

TERMINAL DESCENT PROPULSION SYBSYTEM SUMMARY COMPARISON OF ALTERNATIVES

CANDIDATES	EVAL		
	PROBABILITY OF MISSION SUCCESS	SYSTEM PERFORMANCE	DEVELOPME RISK
Four Engines Configuration	• **More engines, less effect from misalignments** • Roll control by canting engines is possible (2°–10°). • **Requires tight tolerance on ignition delay** • Most compatible with landing radar **(3 or 4 beam)** • **Best engine size/system complexity**	• **Weight** (875 lb) • **Least volume** (11.8 ft^3) • Small exit area (1.97 ft^2) • Small combined area required (5.27 ft^2) • **Short engine (25 in.)** • Small diameter (9.5 in.)	• Simplest sys • No gimbal re • Predicted de years) • Least check and pre-laun
Three Engines Configuration	• Muliple engine misalignments have less severe effect • Simplest • Requires tight tolerance on ignition delay	• Weight (863 lb) • Volume (12.8 ft^3) • Small exit area (1.87 ft^2) • Smallest combined area (5.17 ft^2) • Moderate length (28 in.) • Small diameter (11 in.)	• Simple syste • Requires de gimbal • Predicted de years)
One Engine Configuration	• RCS has one engine out reliability • Reduced reliability of the radar because of split antenna (ΔR = .0012) • Least tip-over at ignition • Longest plume – affects Aeroshell longer and ground sooner • Maximum component reliability • Maximum contamination potential • Least effects on radar • Forces inertial sensors off c.g.	• Lightest system weight (844 lb) • Largest volume (15.3 ft^3) • Minimum engine exit area (1.77 ft^2) • Maximum combined radar and exit area (5.87 ft^2) • Longest engine (38 in.) • Largest diameter (18 in.)	• Requires au capability o • Requires mo level, all m changes • Check of st • Predicted d years) • Requires in antenna
Six Engines Configuration	• Most engines; least effects from misalignments • Provides engine-out capability; needs a failure detection and logic net • Most complex • Requires tight tolerance on ignition delay • Minimal contamination potential	• Smallest diameter (9 in.) • Heaviest (921 lb) • Volume (13.6 ft^3) • Largest exit area (2.64 ft^2) • Largest combined area required (3.94 ft^2) • Shortest engines (19 in.) • Stepped throttling operation is feasible	• Most compl one-engine- detection) • Gimbal requ • Predicted d years) • System test facility req

Figure 4.2–25

UATION CRITERIA			
NT	VERSATILITY	COST	SELECTION
tem so lowest risk quired velopment time (3.5 out time in system test ch	• Most adaptable to lander configuration • Can be modified in thrust level on Capsule design • 10:1 throttling adequate	• Low engine cost because gimbals not needed • Highest cost multiple engine system ($80.5 million) (Difference may be negligible or may reverse order)	X
m so low risk elopment of engine velopment time (3.5	• Adaptable to lander configuration • Low adaptability of roll control owing to engine gimballing • 10:1 throttling provides adequate growth	• Least cost system, at low point between engine costs and system costs ($79.4 million) • High engine unit costs	
iliary attitude control roll control and TVC lification of pressure terial, flow path rilizability velopment time (2.5 tallation of split	• Least adaptable to preferred lander configuration • Control location makes redesign of Capsule the most involved • 10:1 throttling provides adequate growth capability	• Requires development of two engines – main and RCS • Potentially the highest cost ($81.2 – $99.7 million) • Requires split landing radar antenna development	
k system especially if ut capability (failure red velopment time (3.5 most complex and rements most severe	• Step operation capability provides added adaptability • Most adaptability to 1969 data inputs • Easiest to modify thrust level • Step operation could be used to reduce 10:1 continuous throttling, or to improve growth potential	• Requires development of a failure detection and logic net to use 5 out of 6 engine capability • Engine (only) development costs least because they are smallest • Low costs ($79.8 million)	

A schematic diagram for the preferred terminal propulsion approach is shown in Figure 4.2-26.

4.2.2.6 Telecommunications - The primary function of the Capsule Bus telecommunication is the transmission of Capsule Bus engineering data to Earth via the Spacecraft. A second function is to receive commands from the Earth prior to Spacecraft-Capsule Bus separation. A block diagram of our preferred design is shown in Figure 4.2-27. This choice was made after extensive optimization studies and comparative analysis of alternative concepts, as summarized in Figure 4.2-28.

Immediately after separation of the Capsule Bus from the Spacecraft, the radio link to the Spacecraft is put into operation. Two 5 watt transmitters and two spacecraft-mounted receivers with a diversity combiner operate simultaneously in the 300-400 MHz band.

One of the critical problems inherent in the relay communication link is the potentially severe multipath interference. The preferred design minimizes multipath interference by using frequency and time diversity, effective antenna pattern, and Spacecraft/Capsule Bus positions at the time of entry.

It is expected that a blackout period will be encountered during atmospheric entry, and therefore we have incorporated a delay storage memory in the telecommunications link. Each data bit is transmitted in real time, in addition to being retransmitted with 50 second and 150 seocnd time delay. This insures that each bit is transmitted at least once prior to landing, accommodating a range of entry trajectories and resulting blackout intervals.

For extra reliability and improved performance in the presence of multipath interference, the Capsule Bus telemetry system is interleaved with the Entry Science Package telemetry.

Figure 4.2-29 lists the operational modes of the Capsule Bus telecommunications system.

4.2.2.7 Thermal Control - This subsystem maintains an acceptable environment for the Capsule Bus structure and temperature sensitive components during all mission phases. Major elements include a multilayer insulation blanket over the outer canister surface, thermostatically controlled heaters and insulation for components with special requirements, and a thermal curtain over the exposed rear areas of the Aeroshell during entry.

The multilayer insulation blanket is the primary thermal resistance between the Capsule Bus and the deep space environment. It consists of 30 sheets of mylar, each

SCHEMATIC DIAGRAM TERMINAL DESCENT PROPULSION SUBSYSTEM – VOYAGER

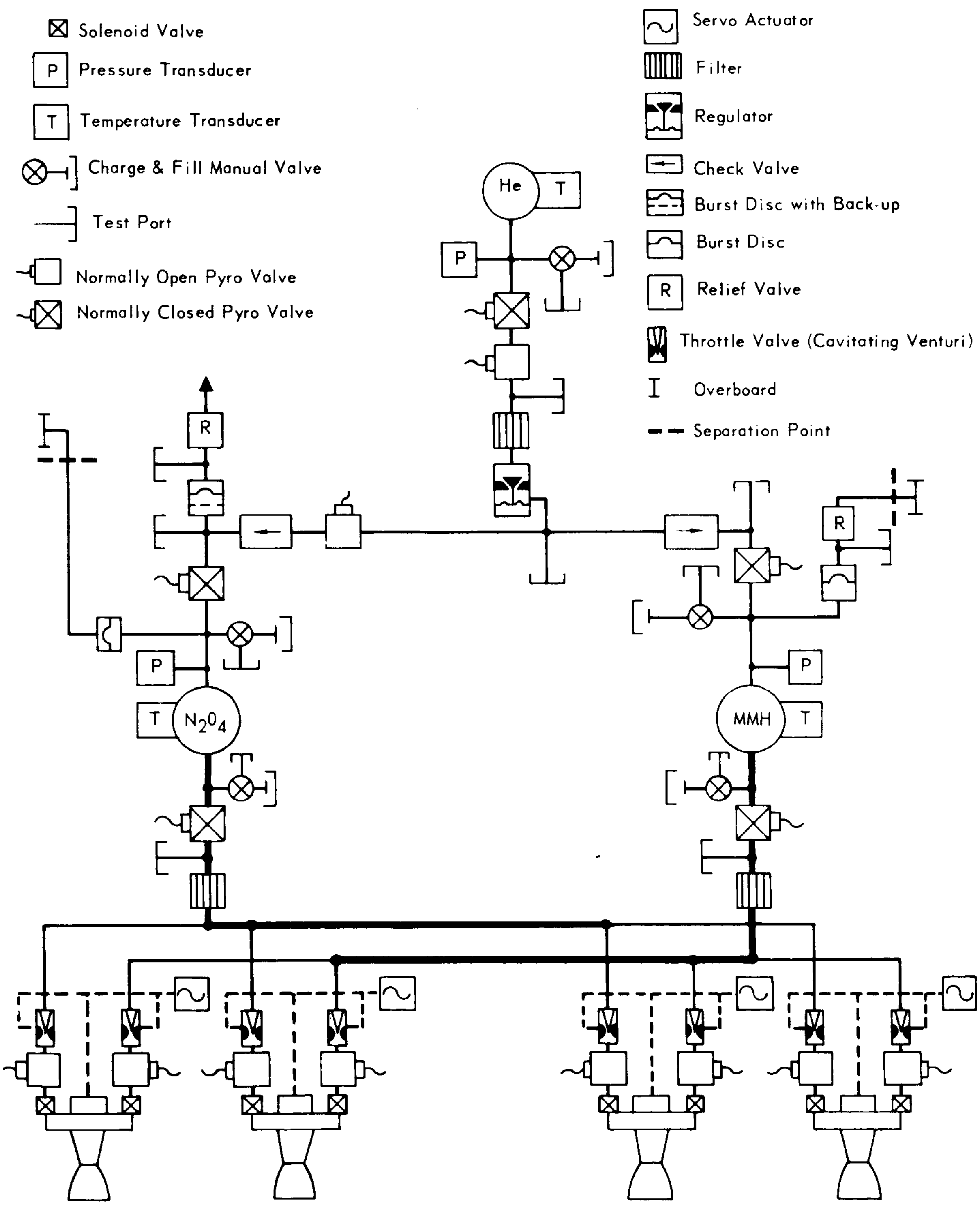


Figure 4.2–26

CAPSULE BUS
TELECOMMUNICATIONS BLOCK DIAGRAM & PERFORMANCE CHARACTERISTICS

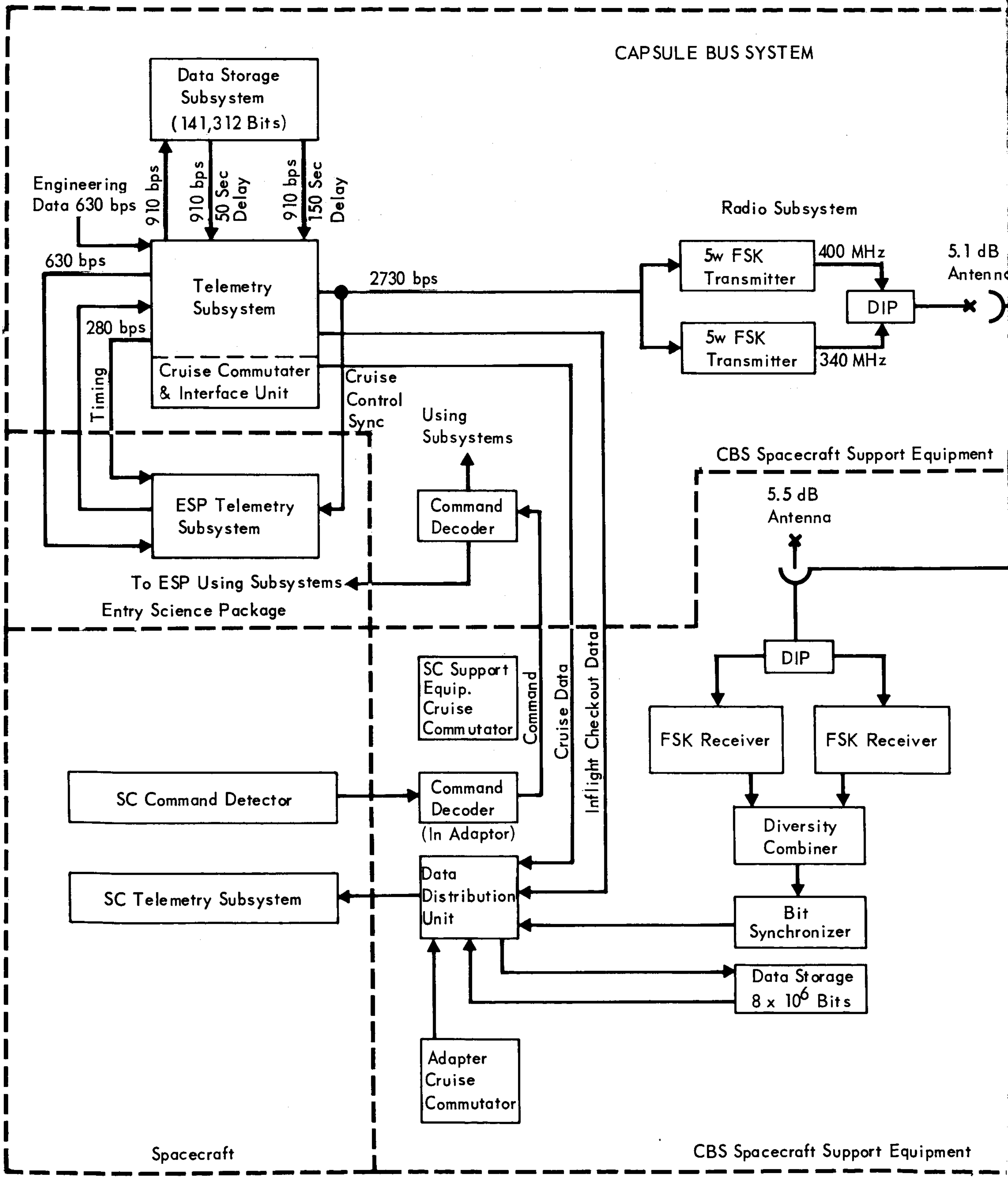
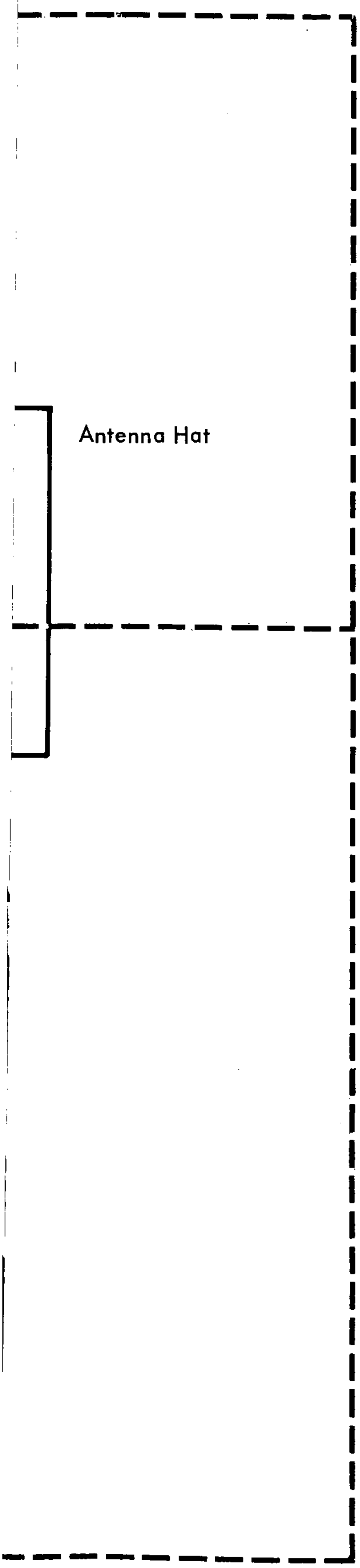


Figure 4.2-27



PERFORMANCE CHARACTERISTICS

Input Data Rate: 630 bps
Output Data Rate: 2730 bps
Data Storage: 50 sec and 150 sec Delay Storage
To provide data accumulated during blackout.
Radio Link:
Modulation: FSK with split phase coding
Frequency Diversity
Carrier Frequencies: 340 MHz and 400 MHz
Transmitter Power: 5 w each
Antennas:
Transmitting: 5.1 dB Cavity–Backed Spiral; 95° Beam
Receiving: 5.4 dB 4-Element Array; 180° x 48° Beam
Redundant Path Alternatives:

- Two paths vis ESP during entry phase
- Degraded performance in multipath environment with only one link of dual radio link operating

CAPSULE BUS
TELECOMMUNICATION OPTIMIZATION STUDY SUMMARY

FUNCTION	CANDIDATES	EVALUATION FACTORS: PROBABILITY OF MISSION SUCCESS	SYSTEM PERFORMANCE	DEVELOPMENT RISK	SELECTION
Configuration	Direct-To-Earth	Poor – Antenna Pointing During Descent	Lower Capacity Due To Space Loss	High – Need For Antenna Development	
	Relay To Spacecraft	**Good**	**Better**	**Low**	x
Modulation	PSK/PM	Poor – Due To Multipath Suscepitibility	Highest Capacity	Low	
	PCM/FM	Good	Lower Capacity	Low	
	MFSK	Good	Higher Capacity	High	
	DPSK	Good	Higher Capacity	High	
	FSK	**Best – Performance More Predictable**	**50% Lower Capacity Than Best**	**Low**	x
Synchronization	Data Channel RZ	Good	Needs More Power	Low	
	Data Channel NRZ	Poor – Due To Difficulty in Sync During Multipath	Best When In Sync	Higher – Need More Complex Sync Circuits	
	Separate Sync Channel	Need Extra Components	Need Extra Power	Low	
	Data Channel Split Phase	**Good – Even In Multipath**	**Almost Equal To NRZ**	**Low**	x
Transmitting Antenna Type	Array	Low-High Complexity	Best – High Gain And Multipath Discrimination	High	
	Fan Beam	Fair – Requires Roll Control	Good – Some Gain And Multi-path Discrimination	Low	
	Broad Beam Conical (Roll Symmetrical)	**Good – No Roll Control Req.**	**Adaquate**	**Low**	x
Black Out Data Recovery	Programmed	Fair	Similar	High	
	Continuous Delay And Interleave	**Good – No External Sensors Or Control Req**		**Low**	x
Multipath Data Recovery	Delay Storage	Poor – Need External Sensors And Complex Storage	Good	High	
	Time Diversity	Poor – Complex Spacecraft Mounted Equip.	Poor – Lack Of Time Before Touchdown	High	
	Frequency Diversity	**Good Simple Receiver**	**Good**	**Low**	x
Spacecraft Data Handling	Relay In Real Time	Good	Best – From Operations Stand-point	Similar	
	Store And Forward	Least – Req Tape Recorder			
	Relay In Real Time, Store And Forward	**Best – Alternate Paths**	**Best**		x

Figure 4.2–28

CAPSULE BUS
TELEMETRY MODES

MISSION PHASE	MODE	COMMENTS
Prelaunch Validation	All	All subsystems up – All modes followed by an "as required" period
Launch thru Interplanetary Cruise	Cruise	Cruise commutator through DDU to S/C TM subsystem. Continuous during launch thru interplanetary trajectory injection.
Midcourse Correction(s)	Test	Fuel gauging mode. Specfic channels monitored and data stored during acceleration periods. Data dump at MOS option after completion of maneuver.
Planetary Orbit Injection	Test	TM up full, data storage during injection maneuver. Calibrate CB gyros, accel., fuel quantity, etc. Data dump at MOS option after injection.
Orbit	Cruise	Cruise Commutator operating.
Preseparation Checkout	Test	All subsystems full up, all operational modes validated. Subsequently, on "as required" basis.
De-orbit and Orbital Descent	Descent	RF to S/C relay terminal
Entry	Entry	Entry mode. Starts at 800K ft. RF to S/C. ESP and SLS data interleaved. Delay storage operating.
Terminal Deceleration	Terminal	Terminal mode. Starts prior to Aeroshell sep. RF to S/C. ESP and SLS data interleaved. Ends after landing.

Figure 4.2–29

using highly reflective aluminum coating. Our studies have indicated that placing the insulation external to the canister surface is the best method for thermal protection of the Capsule Bus, as summarized in Figure 4.2-30.

A thermal curtain is used to protect components and structure within the conical Aeroshell from overheating during the atmospheric entry phase and to reduce heat loss from the rear of the Aeroshell during de-orbit. This curtain is a .03 inch fiberglass cloth. The surface facing the interior of the Aeroshell is gold coated, to minimize radiation. It is removed by being pulled away at parachute jettison prior to landing.

During interplanetary cruise, certain items of temperature-sensitive equipment are provided with individual insulation and electrical resistance heaters (which use power from the Spacecraft solar panels). The selected insulation material is fiberglass with a silicone binder. The equipment which is thermally controlled in this manner is listed in Figure 4.2-31.

4.2.2.8 Electrical Power - The electrical power subsystem consists of one sealed, manually activated silver zinc battery, three automatically activated silver zinc batteries, one battery float charger, two dc-to-dc converter regulators, and one power switching and logic unit (PS&L). A block diagram is shown in Figure 4.2-32. Power for Capsule Bus equipment operation during cruise is provided from the Flight Spacecraft, except during periods of Flight Spacecraft high power usage.

Silver zinc batteries were selected because they are adaptable to heat sterilization. Significant development work directed toward this end has already been accomplished by ESB Corporation, Eagle-Picher, and Douglas Astropower.

The total weight of the electrical power subsystem is 120.5 pounds and it occupies a volume of 2000 cu. in. The physical characteristics of the equipment are shown in Figure 4.2-33.

Redundancy to the Capsule Bus power subsystem is provided by one of the four Surface Laboratory batteries. Automatic voltage sensing is used to bring the Surface Laboratory battery on line if required. We preferred this arrangement, in lieu of providing block redundancy, in the interest of saving total landed weight, even though it creates an interface condition.

EVALUATION OF CAPSULE BUS INSULATION PLACEMENT

INSULATION PLACEMENT	EVA		
	INTERFACE WITH CANISTER SEPARATION TIMING	INSULATION SEPARATION TECHNIQUE	THERMAL CON TECHNIQUE DURIN ORBITAL DESC
Outside canister (attached to canister external surface)	Canister separated in Mars orbit	Separate with canister in single sequence	Solar heat input to hea rolling capsule
Inside canister (attached to canister internal	Canister separated in Mars orbit	Separate with canister in single sequence	Solar heat input to hea rolling capsule
Inside canister (attached to heatshield)	Canister separated during cruise	Separate after canister separation i.e. two separation sequences required	Insulation retained ov shield to prevent ex heat loss

Figure 4.2–30

SUMMARY OF CAPSULE BUS EQUIPMENT REQUIRING THERMAL CONTROL

EQUIPMENT	THERMAL CONTROL METHOD	
	INSULATION	HEATER
RCS Propellant Tanks	X	X
Terminal Descent Propellant Tanks	X	X
De-Orbit Motor		X
Propulsion System Valves	X	X
Canister Power Subsystem	X	X
Canister Sequencing Subsystem	X	X
Canister Telemetry Subsystem	X	X

Figure 4.2–31

...UATION FACTORS				
...TROL ...G MARS ...ENT	LANDING SITE CONSTRAINT	ENVIRONMENTAL EFFECTS ON INSULATION	SUBSYSTEM INTERFACES	SELECTION
...tshield,	Land between the morning terminator and noon	Possible micrometeoroid and ground handling damage	Mechanical interface with canister; Canister structure and separation devices not exposed to cold space environment	X
...tshield,	Land between the morning terminator and noon	Possible damage during terminal sterilization	Mechanical interface with canister	
...er heat- ...cessive	No thermal control constraint	Possible damage during terminal sterilization	Windows required in insulation for sensors since aluminized Mylar not RF transparent	

CAPSULE BUS ELECTRICAL POWER SYSTEM BLOCK DIAGRAM

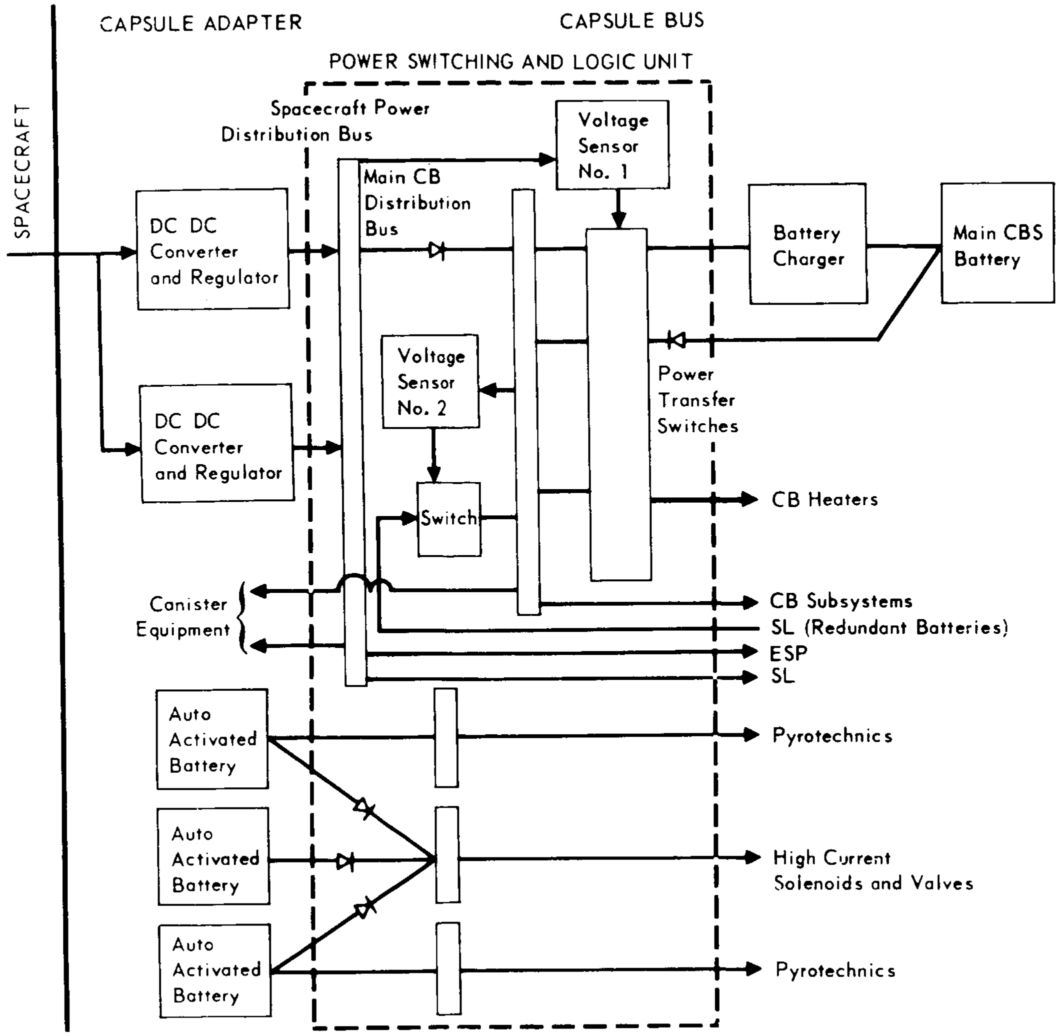


Figure 4.2–32

CAPSULE BUS
ELECTRICAL POWER SUBSYSTEM PERFORMANCE CHARACTERISTICS

ITEM	PERFORMANCE CHARACTERISTICS				
	TYPE	WEIGHT (lb)	QUANTITY	CELLS/BATTERY	ELECTRICAL
Main CB Battery	Sealed silver zinc	64	1	19	Energy 1700 W–hrs
Auto Activated Battery	Sealed silver zinc	8	3	22	Peak discharge current 20 amps each
DC to DC Converter		4	2	–	Power output 250w
Battery Charger	Float Charger	3	1	–	–
Power Switching Logic Unit		11	1	–	–

Figure 4.2–33

4.3 ENTRY SCIENCE PACKAGE - The objectives of the Entry Science Package (ESP) are to measure the atmospheric properties of Mars and to obtain high resolution images of its surface.

We have complied with the requirement of the VOYAGER constraints document to design the Entry Science Package as an independent system to the maximum extent practicable. However, even with this goal in mind, the inherent interfaces between the Entry Science Package and the Capsule Bus are such as to result in considerable interdependence between these systems, both from a physical and an operational standpoint. (This is less true in the ESP interfaces with the Surface Laboratory.) Another important factor is that the design requirements for the supporting subsystems of the Entry Science Package and the Capsule Bus have an inherent commonality.

Therefore, we recommend that the technical and management responsibility for the Entry Science Package be placed with the Capsule Bus System Contractor. This will lead to synergistic benefits in interface simplification and more efficient functional integration.

The principal events affecting Entry Science Package operation are shown in Figure 4.3-1. Transmission of data from all science instruments - except the data from the stagnation point temperature probe and the mass spectrometer - is initiated at 800,000 feet. Operation of these two instruments starts after the point of peak dynamic pressure is passed, programmed to approximate Mach 5. TV operation is continuous until shortly before touchdown, with a 5 second interval between images from alternate cameras. Data from the science instruments, as well as low bit rate engineering data, are stored for a delayed transmission, to avoid the anticipated periods of communications blackout. TV images obtained during the communication blackout are not stored, however, since the data storage provided for delayed retransmission is limited. Stagnation pressure and temperature transducers are separated with the Aeroshell. Base region temperature and pressure, along with atmospheric composition data, continue to be transmitted, however, until after touchdown.

The arrangement of the Entry Science Package within the capsule is shown in Figure 4.3-2. The subsystems that specifically support the experiments have been packaged in a self-contained, independent module, located on the lander base platform adjacent to the Surface Laboratory. Two TV cameras are attached to the footpad of the landing impact attenuation system and view through a single fused silica window on the conical section of the Aeroshell, just aft of the spherical nose cap. The stagnation point pressure and total temperature transducers are located in the nose cap of the Aeroshell, behind a beryllium heat sink plug. Other sensors are located as shown in the figure.

MISSION PROFILE FOR ENTRY SCIENCE PACKAGE

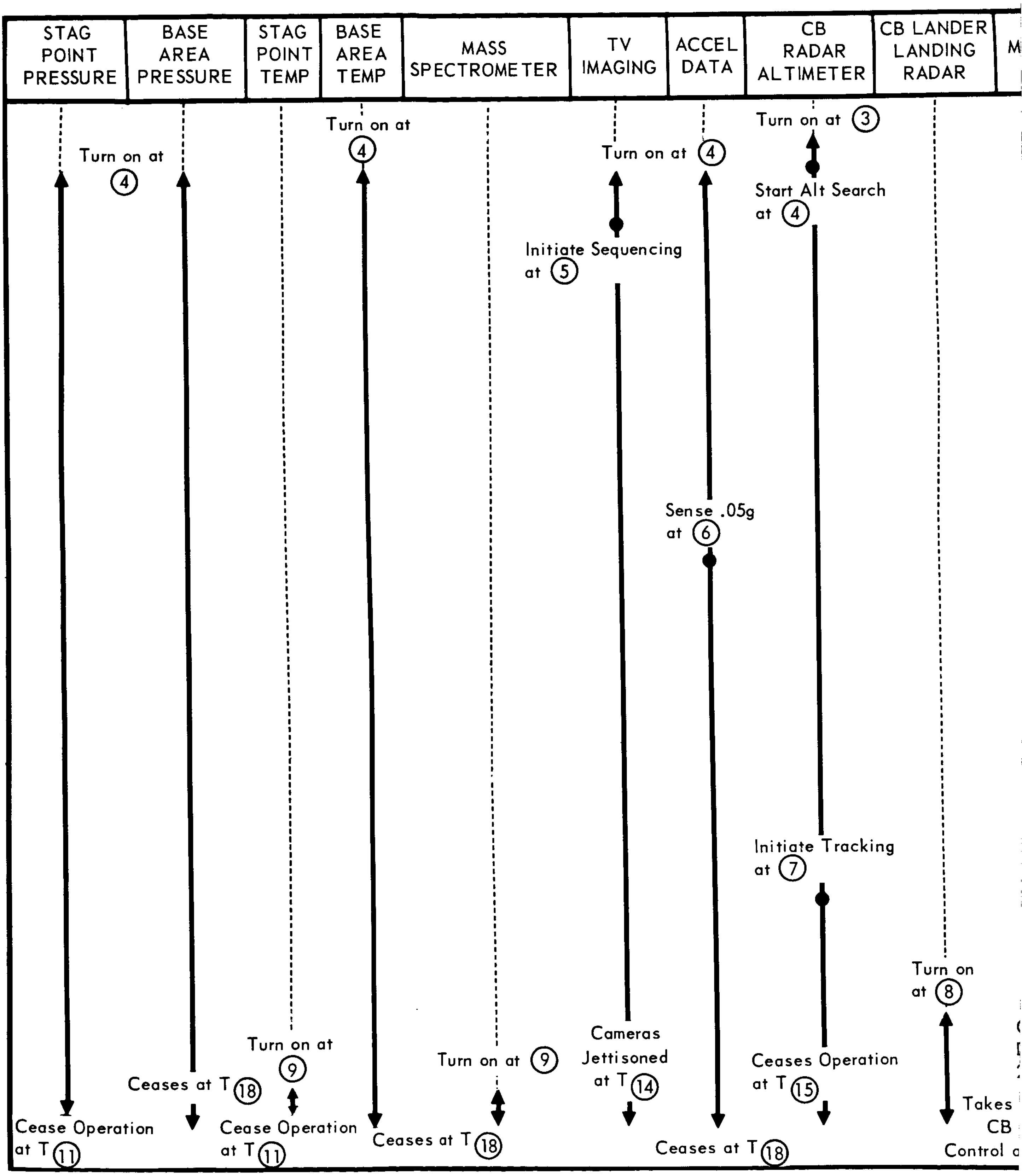
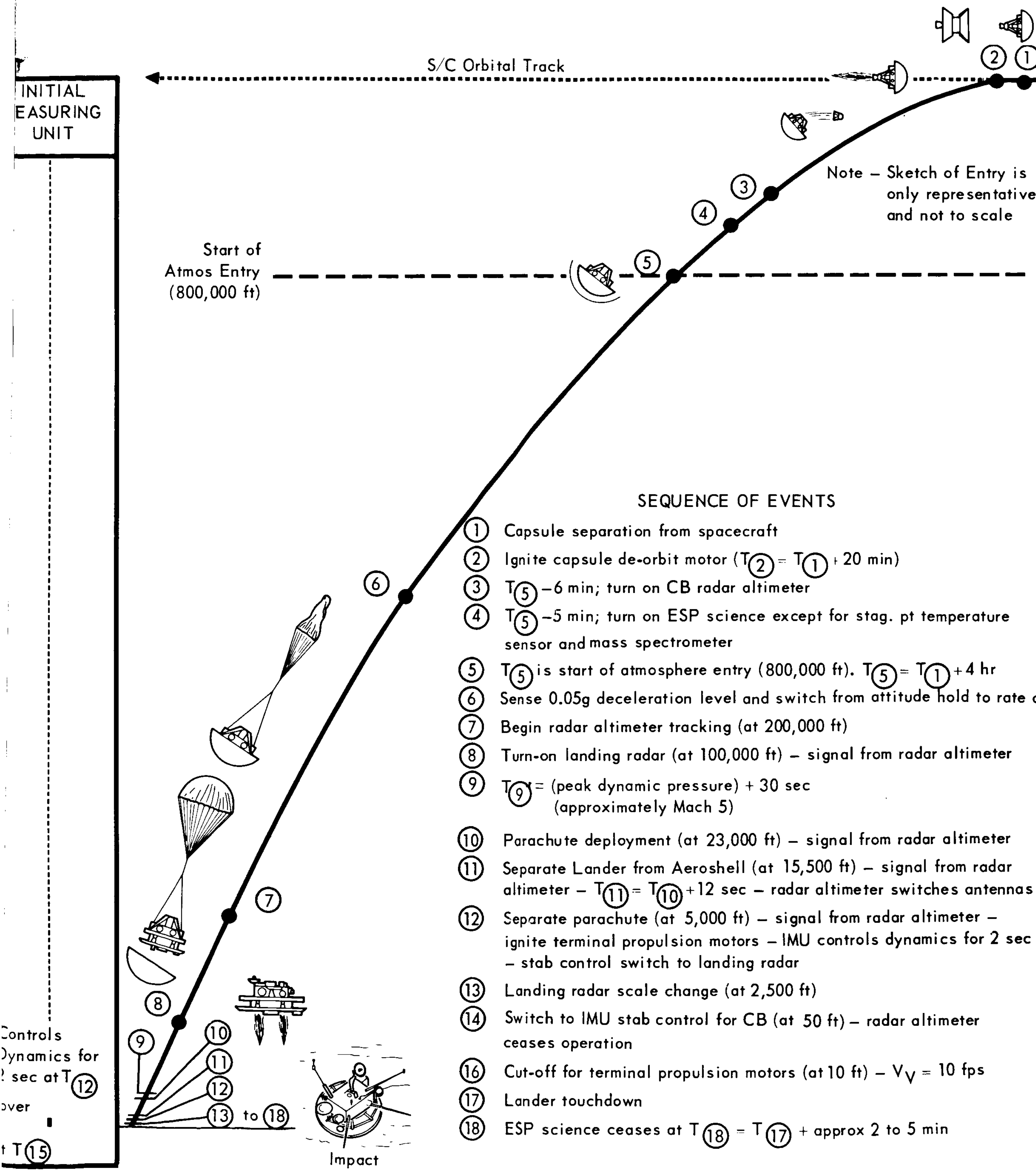


Figure 4.3–1

4–52 –1



SEQUENCE OF EVENTS

1. Capsule separation from spacecraft
2. Ignite capsule de-orbit motor ($T_2 = T_1 + 20$ min)
3. $T_5 - 6$ min; turn on CB radar altimeter
4. $T_5 - 5$ min; turn on ESP science except for stag. pt temperature sensor and mass spectrometer
5. T_5 is start of atmosphere entry (800,000 ft). $T_5 = T_1 + 4$ hr
6. Sense 0.05g deceleration level and switch from attitude hold to rate damping
7. Begin radar altimeter tracking (at 200,000 ft)
8. Turn-on landing radar (at 100,000 ft) – signal from radar altimeter
9. T_9 = (peak dynamic pressure) + 30 sec (approximately Mach 5)
10. Parachute deployment (at 23,000 ft) – signal from radar altimeter
11. Separate Lander from Aeroshell (at 15,500 ft) – signal from radar altimeter – $T_{11} = T_{10} + 12$ sec – radar altimeter switches antennas
12. Separate parachute (at 5,000 ft) – signal from radar altimeter – ignite terminal propulsion motors – IMU controls dynamics for 2 sec – stab control switch to landing radar
13. Landing radar scale change (at 2,500 ft)
14. Switch to IMU stab control for CB (at 50 ft) – radar altimeter ceases operation
16. Cut-off for terminal propulsion motors (at 10 ft) – $V_V = 10$ fps
17. Lander touchdown
18. ESP science ceases at $T_{18} = T_{17}$ + approx 2 to 5 min

ENTRY SCIENCE PACKAGE INSTALLATION

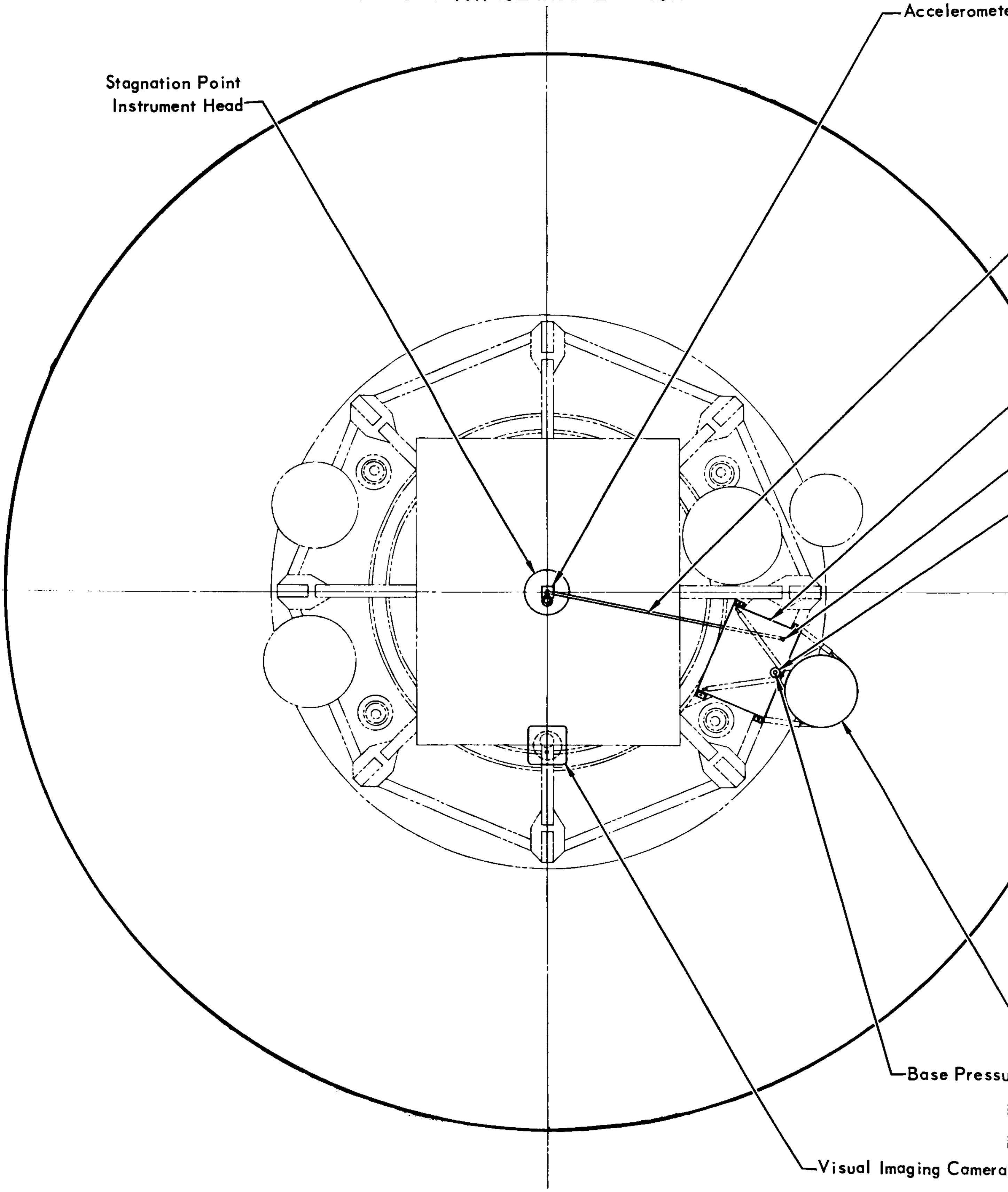
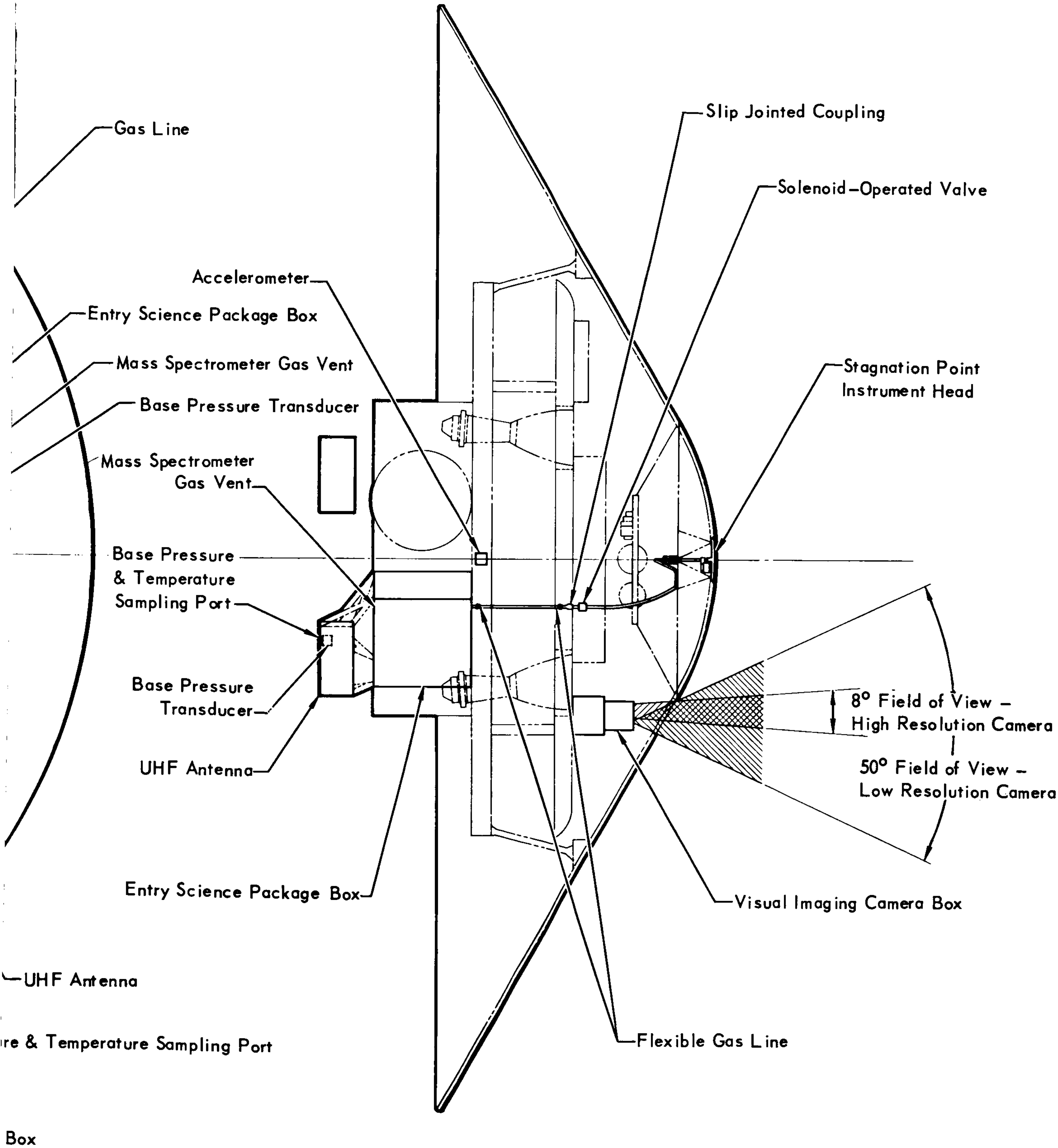


Figure 4.3-2


Gas Line
Slip Jointed Coupling
Solenoid-Operated Valve
Accelerometer
Entry Science Package Box
Mass Spectrometer Gas Vent
Base Pressure Transducer
Stagnation Point
Instrument Head
Mass Spectrometer
Gas Vent
Base Pressure
& Temperature
Sampling Port
Base Pressure
Transducer
UHF Antenna
8° Field of View –
High Resolution Camera
50° Field of View –
Low Resolution Camera
Entry Science Package Box
Visual Imaging Camera Box
UHF Antenna
ire & Temperature Sampling Port
Flexible Gas Line
Box

A summary weight statement is given in Figure 4.3-3. As noted, the instrument list provided by the constraints document - consisting of 27 pounds of scientific instruments - is supported by an additional 173 pounds of equipment, of which about 19 pounds is installed in the spacecraft as part of the relay communications link.

4.3.1 Instruments - The pertinent characteristics of the science instruments included in the preferred design are given in Figure 4.3-4. They are compatible with those specified in the VOYAGER constraints document. Potential high value candidates for use in future missions are:

a. Measurement of γ backscatter from outside the shock wave, for direct determination of the density of the atmosphere.

b. Measurement of solar UV and X-ray radiation absorption during entry, for obtaining supplemental atmospheric density and composition data.

c. Mass spectrometer measurement of atmospheric composition at altitudes above the high aerodynamic heating zone but within the gas continuum region.

d. Post-touchdown imaging by means of a facsimile camera provided in the Entry Science Package and erected after touchdown. This would utilize the ESP/Spacecraft relay link and serve as a backup to the Surface Laboratory imaging experiment.

e. Measurement of differential pressure between the stagnation point and another point on the side of the spherical nose section, for supplemental determination of dynamic pressure independent of the lift and drag characteristics of the Capsule Bus.

A location arrangement for sensors for these added priority measurements is shown in Figure 4.3-5.

4.3.2 Experiments - The instruments for the ESP are used for three types of scientific experiments. These are: imaging, atmospheric density and temperature profile determination, and atmospheric composition determination.

Imaging - The major factors influencing the descent imaging experiment for VOYAGER are listed in Figure 4.3-6. One of the more important aspects is the time sequence of images prescribed. We have used continuous uniform sequencing with a 5 second interval between alternate images from each of the two vidicon cameras. The purpose is to obtain good identification continuity between images, with some stereo overlap, and to assure a continuous transmission bit rate. Unobstructed viewing is another problem. Our design locates the camera window just aft of the non-ablative ceramic nose cap in order to eliminate the effects of ablative gas overflow and its potential condensation on the cooler window.

ENTRY PACKAGE GROUP WEIGHT SUMMARY

	BEFORE AEROSHELL SEPARATION	AFTER AEROSHELL SEPARATION
Structure	14.3	14.3
Thermal Control	5.0	5.0
Telecommunications	55.0	55.0
Electrical Power	22.5	22.5
Experiments	27.0	25.5
Wiring and Mounting Provisions	56.8	55.7
Total Entry Package Weight	180.6*	178.0

* 19 lb of ESP Equipment is part of the 50 lb capsule bus – Spacecraft mounted equipment allocation.

Figure 4.3–3

PREFERRED ENTRY SCIENCE INSTRUMENTS CHARACTERISTICS SUMMARY

	WEIGHT (LB)	VOLUME (In^3)	POWER (WATTS)	MAXIMUM ENERGY (WATT-HRS)	TOTAL DATA (KILOBITS)
Stagnation Region Pressure Transducer	1.0	6.3	1.4	0.35	9.7
Stagnation Temperature Transducer	0.5	1.7	0.01	0.0025	8.7
Base Region Pressure Transducer	1.0	6.3	1.4	0.42	8.7
Base Region Temperature Transducer	0.5	1.7	0.01	0.003	8.7
Accelerometer	2.0	9.6	4.0	1.2	65.4
Mass Spectrometer	8.0	200	7.0	2.1	47.5
Vidicon Cameras	14.0	700	20.0	5.4	<45,600 (240/Frame)
Totals	270.0	925.6	33.8	9.48	<45,750

Figure 4.3–4

PROVISIONS FOR ADDITIONAL ESP MEASUREMENTS

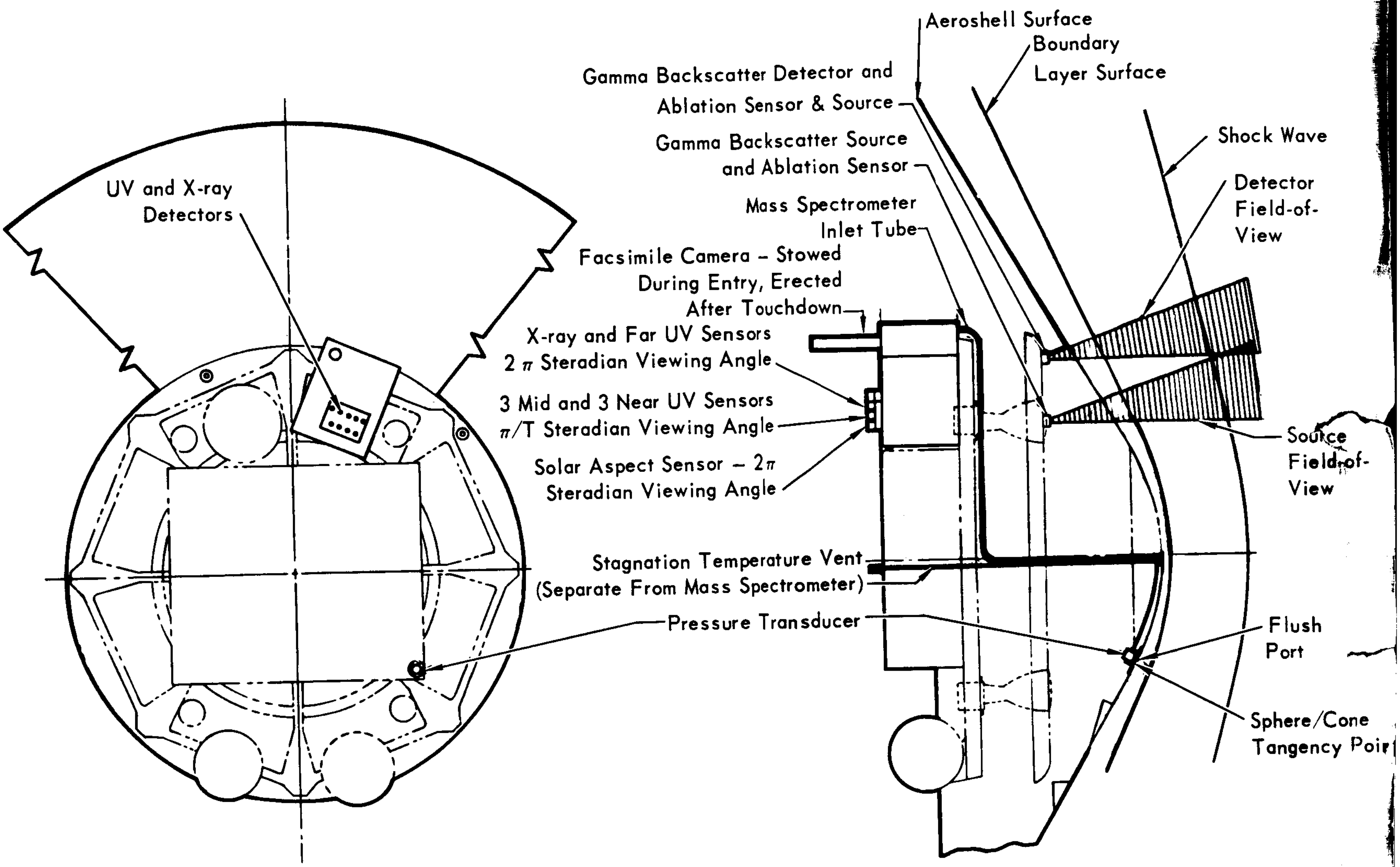


Figure 4.3-5

FACTORS AFFECTING DESCENT IMAGING

OBJECTIVES	ALTITUDE REGIME	CONSTRAINTS	TECHNIQUES
Planetary Limb Photometry	800,000 – 100,000 ft	• Variable Solar Angle • Flow Field Emission	Continuous image sequence from start of descent; non-ablative nose cap.
Terrain Survey	800,000 – 5,000 ft	• Communications blackout • Entry Attitude Variation	Pictures every 5 seconds with a large amount of frame-to-frame overlap giving good correlation a stereoscopic coverage.
Landing Site Location	500,000 –50,000 ft	• Ablative Deposition • Site Visibility	Aeroshell viewing window positioned next to nonablative nose cap; use of two magnifications/fields of view (8° and 50°) to insure 1000 foot ground resolution of site region.
Detailed Surface Study	18,000 – 90 ft	• Aeroshell separation • SLS Impact Safety • Wind Shear Oscillation • Descent Engine Gas Plume • Descent Engine Failure	Camera package mounted to capsule impact pad and pyrotechnically removed at 90 feet; adequate imagery before descent engine ignition of 5000 ft and ability to transmit better than 1 meter ground resolution data if engine fails.

Figure 4.3–6

Atmospheric Profile - Figure 4.3-7 summarizes the inter-relationship between atmospheric sensor instruments and the Capsule characteristics. Strong interaction exists between the local flow field around the capsule and the pressure and temperature sensor outputs. Our approach was to locate one set of sensors at the nose of the Aeroshell and another at the base, in order to get two independent and complementary measurements.

The pressure and temperature data are combined with vehicle velocity and descent altitude - derived from measurements of accelerometers located at or near the c.g. and the Capsule Bus radar altimeter - to reconstruct the atmospheric profile. Errors are reduced when these data are supplemented by measurements of the initial de-orbit conditions and the landing altitude.

Atmospheric Composition - The instrument used to measure atmospheric composition is the mass spectrometer. The inlet port location, line flow, and ionization chamber pressure are quite critical if suitable samples are to be obtained for accurate definition of atmospheric properties with a reasonably small time delay. Aspects of these considerations are also summarized in Figure 4.3-7. Our preferred design places the sampling port at the stagnation point of the Aeroshell. The use of a non-ablative nose cap simplifies the sampling problem. The preferred operational procedure is to conduct this experiment only below Mach 5, to ensure that gases entering the inlet port are not dissociated.

4.3.3 Major Support Subsystems - Certain requirements of subsystem support for the Entry Science Package are similar to those for the Capsule Bus; i.e., portions of telecommunications, power, thermal control, etc. In these cases, we have chosen, wherever practicable, common approaches for implementing these functions, to take advantage of the potential cost saving and development efficiency inherent in commonality.

4.3.3.1 Telecommunications - The Entry Science Package telecommunications subsystem is composed of the following elements:

a. Telemetry

b. Radio

c. Antennas

d. Data Storage

Figure 4.3-8 presents a functional block diagram of the preferred approach, along with its performance characteristics.

ATMOSPHERIC SENSORS – CAPSULE BUS INTERRELATIONSHIPS

INSTRUMENT	DATA CORRECTION/ INTERPRETATION	FIRST STEP DATA UTILIZATION	SUPPLEMENTAL DATA FROM CB EQUIPMENT	HELPFUL CAPSULE BUS CHARACTERISTICS
Accelerometer	• Need CG location and attitude rate for correcting instrument outputs to body axis. • Need aerodynamic coefficients for transfer to flight axes.	• Need aerodynamic coefficients and vehicle mass properties for atmospheric density deduction from acceleration and velocity.	• Rate gyro outputs. • Monitored de-orbit attitude and ΔV for improved entry condition definition. • Altitude measurements before parachute deployment (high altitude altimeter). High altitude measurements facilitate trajectory reconstruction. • RCS duty cycle.	• Accurate entry attitude control. • Rate damping • Access to CG on structural element. • Low aeroelastic vibration at frequencies of interest.
Pressure-Temperature Sensors	• Relationship between free stream dynamic pressure and increment of total pressure/temperature measurements. • Angle of attack - stagnation point relationship.	• Comparison of measured values and those computed with the estimated trajectory and density profile.	• Monitor rate gyro for reconstructing $(\theta - \gamma)$ time history.	• Availability of sensor locations at stagnation point, side of spherical nose cap and base region. • Cleanliness of stagnation influence gas (nonablative nose cap). • Time available below Mach 5 before a) Aeroshell separation b) Terminal propulsion ignition.
Mass Spectrometer	• For possible extension into region of significant dissociation ionization, need prediction of recombination along sampling tube.	• Determination of gas specific heat ratio for pressure/temperature reconstruction.	• Timed functions from peak axial acceleration for measurement initiation. • Time periods of terminal propulsion thrusting.	• Sensor port access in non-ablating nose cap to ensure uncontaminated gas samples. • Time available at low speeds and altitude prior to thruster ignition.

Figure 4.3-7

ENTRY SCIENCE PACKAGE
TELECOMMUNICATIONS BLOCK DIAGRAM AND PERFORMANCE CHARACTERISTICS

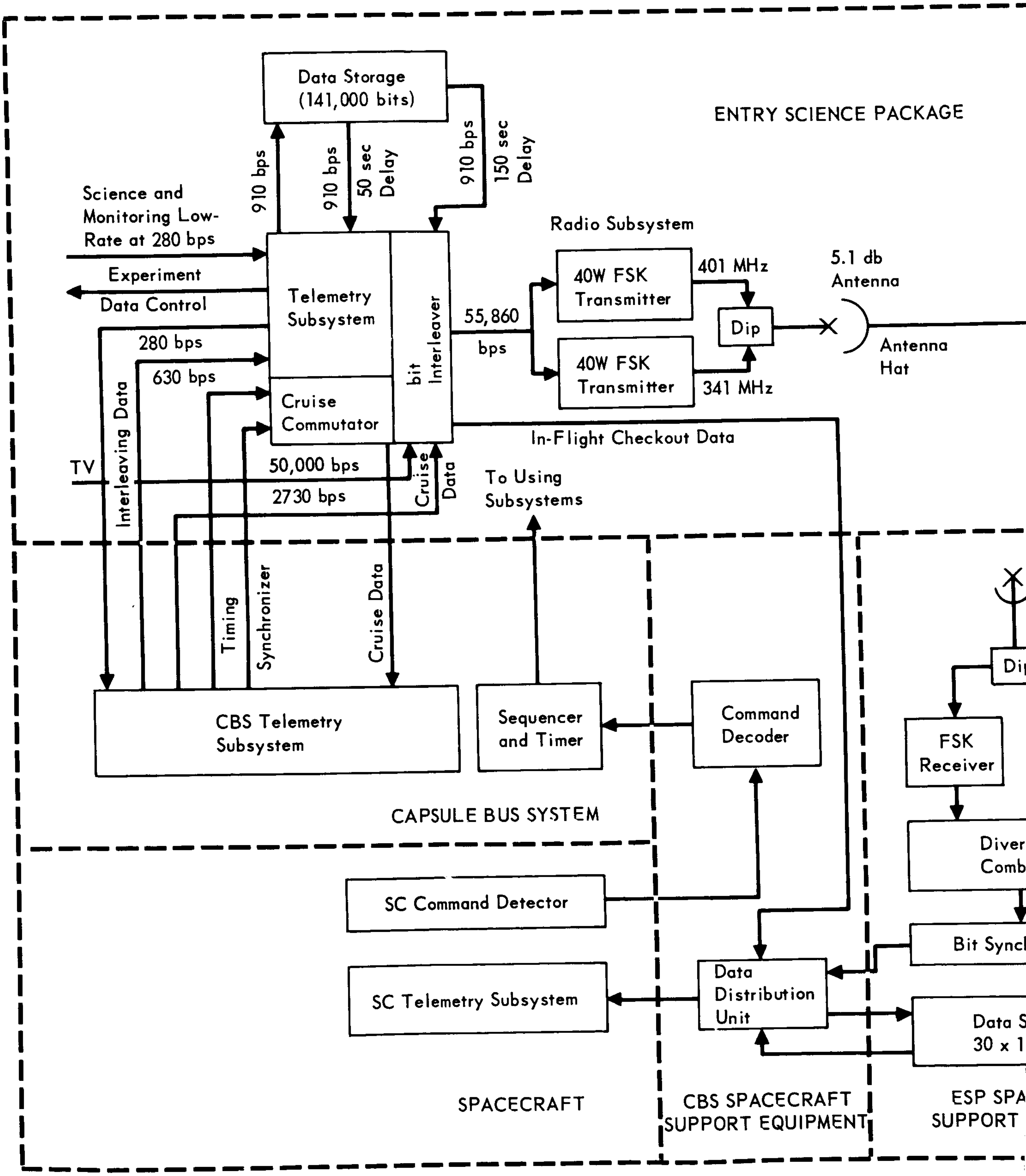
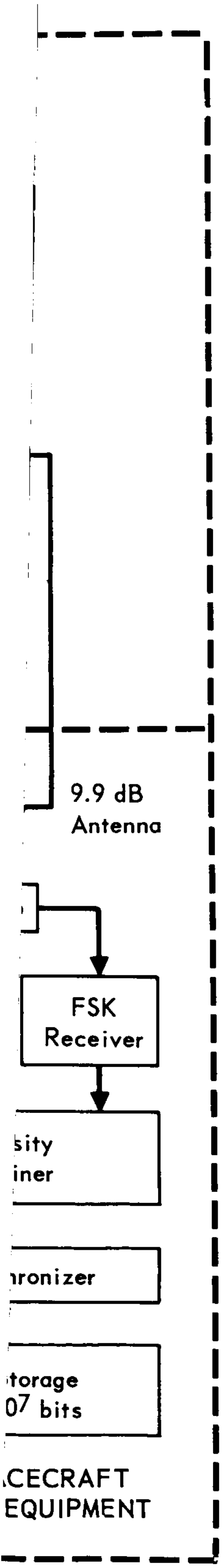


Figure 4.3-8



INPUT DATA RATE:

- Low Rate Science and Engineering: 208 bps
- High Rate Science (TV) 50,000 bps

OUTPUT DATA RATE: 55,860 bps

DATA STORAGE: 50 sec and 150 sec delay storage to provide low rate science and engineering data accumulated during blackout.

RADIO LINK:

- Modulation: FSK with split-phase coding
- Frequency Diversity:
- Carrier Frequency: 341 MHz and 401 MHz
- Transmitter Power: 40W Each

ANTENNAS:

- Transmitting: 5.1 db cavity-backed spiral; 95° beam
- Receiving: 9.9 db single helix; 55° beam

TELEMETERY PROGRAMMER:

Reprogrammable by command prior to separation from spacecraft

REDUNDANT PATH ALTERNATIVES:

- Low Rate Science and Engineering: Through CBS link
- High Rate Science (TV): Available via single link of dual radio link with less multipath margin

Stored programs are used to control the telemetry equipment, thus providing flexibility to accommodate changes in scientific payload data. We have chosen a core memory because of its inherently greater reliability compared to other concepts. All data, except TV, are stored in the core memory and read out 50 and 150 seconds later, to counteract transmission blackout. The resulting data stream simultaneously modulates two solid-state 40 watt FSK transmitters that operate at frequencies of 341 MHz and 401 MHz. The transmitter outputs are combined by a diplexer and radiated by a single-element cavity-backed spiral antenna. The spacecraft receiving antenna is a single axial-mode helix, mounted on a mast. Diversity FSK receivers are utilized in the spacecraft to give improved performance under multipath conditions. Either transmitter-receiver pair operating alone will provide adequate performance, except during periods of severe multipath interference. This feature therefore provides redundancy for increased reliability.

A tape recorder with a 30 million bit capacity is used to store the data in the Spacecraft until transmitted to Earth.

Additional command and telemetry functions are performed prior to capsule-spacecraft separation by special purpose equipment incorporated within the Entry Science Package and Capsule Bus telemetry subsystems. Figure 4.3-9 summarizes the telemetry modes.

Alternate path transmission of all Entry Science Package data (except television) is provided through the Capsule Bus radio subsystem, in order to achieve greater reliability and to minimize the influence of multipath interference.

Figure 4.3-10 summarizes the alternatives considered in the selection of our preferred approach and gives the reasons for the choice.

4.3.3.2 Power - As in the case of the Capsule Bus, our preferred approach for the Entry Science Package is to use a sealed, sterilizable, silver-zinc battery as the power source. The electrical power subsystem provides power for in-flight monitoring during the cruise period when Flight Spacecraft power is not available, and for operation of equipment from pre-separation to a few minutes after landing. The subsystem consists of a battery, a battery charger, and a power switching and logic unit. The battery is designed for a high discharge rate and has 8.5 amp-hour capacity. The battery charge is by a two-step float charger.

In arriving at our preferred approach, we analyzed the desirability of providing the Entry Science Package power directly from the Capsule power system, but decided upon a separate power source mainly to achieve a simpler interface. Because of the importance of a reliabile power system to successful Entry Science Package

ENTRY SCIENCE PACKAGE TELEMETRY MODES

MISSION PHASE	MODE	COMMENTS
Prelaunch Validation	All	All subsystems up – All modes followed by an "as required" period.
Launch thru Preseparation Checkout	Cruise	Cruise commutator through CBS and DDU to SC TM subsystem. Continuous during launch to preseparation checkout.
Preseparation Checkout	Test	All subsystems full up, all operational modes validated. Subsequently, on "as required" basis.
Separation through Orbital Descent	Cruise	Cruise commutator thru CBS telemetry to Spacecraft via CBS–SCS relay link.
Entry	Entry	Entry mode. Starts at 800,000 ft. Relay link to SC. ESP and SLS data interleaved. Delay storage operating, for low rate data.
Terminal Deceleration	Terminal	Terminal mode. Starts prior to Aeroshell separation RF to S/C. ESP and SLS data interleaved. Ends 5 minutes after landing.

Figure 4.3–9

ENTRY SCIENCE PACKAGE
TELECOMMUNICATION OPTIMIZATION STUDY SUMMARY

FUNCTION	CANDIDATES	EVALUATION FACTORS: PROBABILITY OF MISSION SUCCESS	SYSTEM PERFORMANCE	DEVELOPMENT RISK	SELECTION
Configuration	Direct–To–Earth	Poor – Antenna Pointing During Descent	Lower Capacity Due To Space Loss	High – Need For Antenna Development	
	Relay To Space-craft	Good	Better	Low	x
Modulation	PSK/FM	Poor – Due To Multipath Suscepitibility	Highest Capacity	Low	
	PCM/FM	Good	Lower Capacity	Low	
	MFSK	Good	Higher Capacity	High	
	DPSK	Good	Higher Capacity	High	
	FSK	Best – Performance More Predictable	50% Lower Capacity Than Best	Low	x
Synchronization	Data Channel RZ	Good	Needs More Power	Low	
	Data Channel NRZ	Poor – Due To Difficulty in Sync During Multipath	Best When In Sync	Higher – Need More Complex Sync Circuits	
	Separate Sync Channel	Need Extra Components	Need Extra Power	Low	
	Data Channel Split Phase	Good – Even In Multipath	Almost Equal To NRZ	Low	x
Transmitting Antenna Type	Array	Poor – High Com-plexity	Best – High Gain & Multipath Discrimination	High	
	Fan Beam	Fair – Requires Roll Control	Good – Some Gain & Multipath Discrimi-nation	Low	
	Broad Beam Conical (Roll Synmetrical)	Good – No Roll Control Req.	Adaquate	Low	x
Black Out Data Recovery	Programmed	Fair	Similar	High	
	Continuous Delay And Interleave	Good – No External Sensors Or Control Req.		Low	x
Multipath Data Recovery	Delay Storage	Poor – Need External Sensors & Complex Storage	Good	High	
	Time Diversity	Poor – Complex Spacecraft Mounted Equipment	Poor – Lack Of Time Before Touchdown	High	
	Frequency Diversity	Good Simple Re-ceiver	Good	Low	x
Transmitting Antenna Configuration	Share CB Antenna	Good	Poor – Losses In Diplexer Versatility Poor	Similar	
	Have Separate Antenna	Good	Good Best On Versatility Separate Interfaces		x

Figure 4.3–10

operation, we have provided redundant battery capacity via the Surface Laboratory power source. Though this makes a more complex interface, the benefits of the redundancy were overriding. Figure 4.3-11 presents presents a block diagram of the preferred power subsystem.

4.3.3.3 Thermal Control - The thermal control subsystem maintains equipment temperature levels within their allowable ranges throughout the mission. Temperature control is provided for both the science instruments and subsystems and for all Capsule Bus-mounted equipment. Until Capsule Lander separation from the Aeroshell, the subsystem operates within the overall temperature environment provided by the Capsule Bus thermal control subsystem, which averages -140°F prior to entry and has local areas up to 800°F at the time of Aeroshell separation. In the Entry Science Package, however, the equipment temperature is maintained between 50° to 125°F throughout the mission. The major elements of the system include electrical heaters, insulation and thermal control surfaces.

Small heaters are provided in the insulated support equipment module, the descent TV, and the stagnation pressure transducer. The glass fiber insulation minimizes the heating power requirements; prevents excessive cool-down of equipment when power interruptions occur during Spacecraft midcourse corrections; and avoids overheating of equipment during the brief Martian entry period.

4.3.4 Capsule Bus/Entry Science Package Accommodations - Our preferred Capsule Bus design accommodates the special requirements of the Entry Science Package. Some of these design accommodations are listed in Figure 4.3-12. The most outstanding feature is the special non-ablative design of the Aeroshell nose cap - provided to enhance the achievement of the science objectives of the mission, although it resulted in additional weight and fabrication complexity for the Capsule Bus.

ENTRY SCIENCE PACKAGE ELECTRICAL POWER SUBSYSTEM BLOCK DIAGRAM

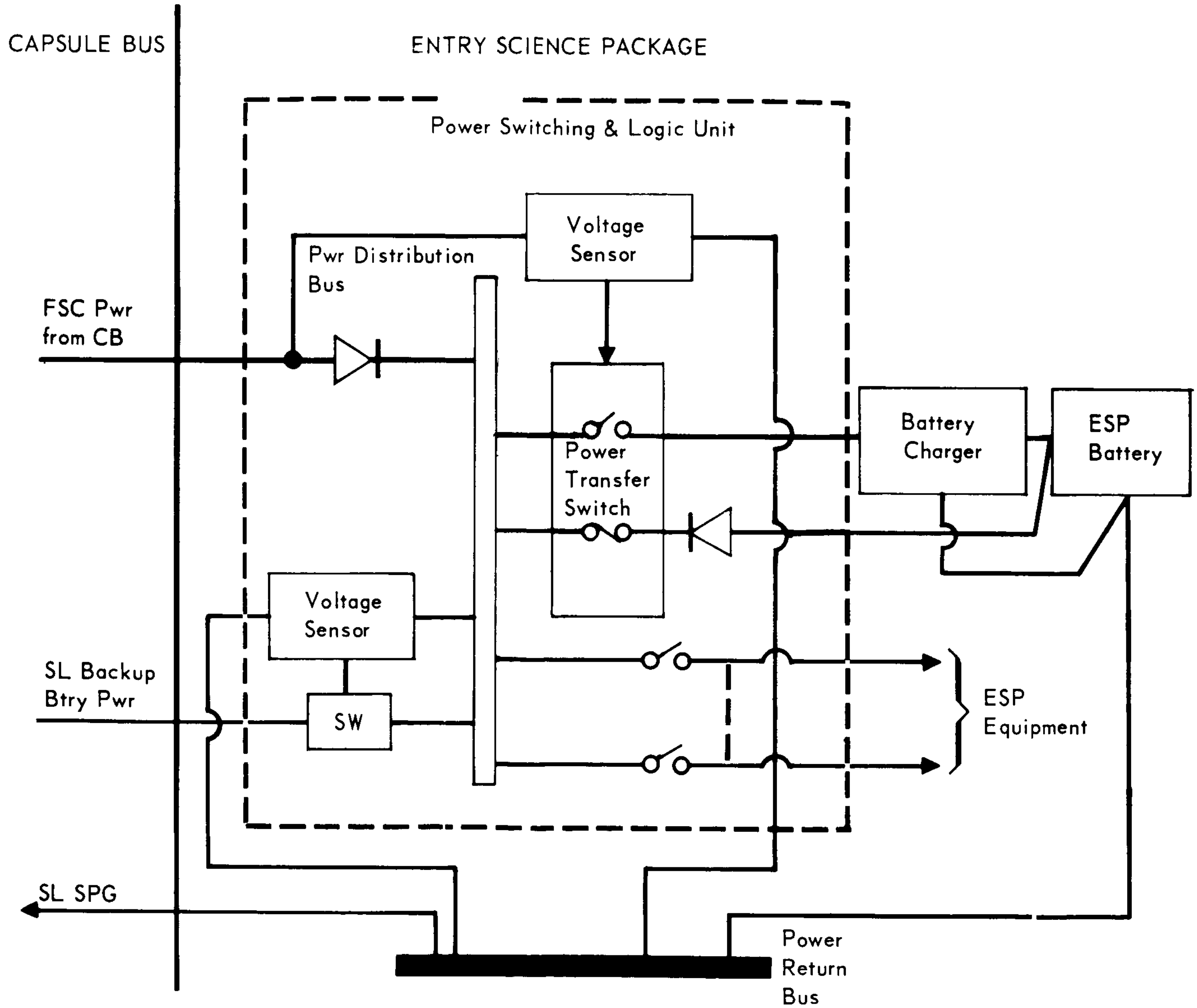


Figure 4.3-11

4-64 – 1

CAPSULE BUS AND ENTRY SCIENCE PACKAGE MUTUAL ACCOMMODATION

– Capsule bus requirements established solely by ESP

* – Indentifies items influenced by ESP requirements.

ESP FUNCTION	CAPSULE BUS DESIGN OR OPERATIONAL FEATURE	SELECTED ESP ACCOMMODATION OR EFFECT	ADDITIONAL CHARACTERISTIC OF FEATURE/ACCOMMODATION
Imaging	• No roll attitude control	• Image camera optically aligned with roll axis	• No roll attitude reference required.
	• Non-ablative nose cap	• Single fuzed quartz window adjacent to non-ablative spherical nose cap	• Good viewing with minimum programming of images
	• Four Terminal Descent Engines	• Camera location to be maximum possible distance from engines compatible with above window location.	• Minimum optical disturbance by exhaust plume during terminal descent.
	• Attitude Rate Damping*	• Improvement in image resolution.	
Pressure, Temperature and Composition	• Beryllium Nose Plug at stagnation point in non-ablative nose cap	• Stagnation point instrument head, ports flush with Aeroshell mold line.	• Composition contamination avoided. • Beryllium nose plug acts as a heat sink for the temperature measurement plus provide structural support for the instrument head. • System Test access to instruments.
	• Attitude Rate Damping*	• Increased accuracy of stagnation point measurements with low α • Uncomplicated mathematical relationship of stagnation measured data to free stream data	
	• Base region thermal curtain with provision for protruding [illegible]*	• Base region pressure and temperature sensors mounted to principal unit structural unit.	
Acceleration	• Parachute catapult along roll axis.	• Mount accelerometers on mortar base structure fitting as close to center of gravity as physically possible.	• Minimize acceleration errors due to center of gravity location.
	• Attitude rate damping*	• Reduce effect of uncertainty in acceleration corrections and $C_L(\alpha)$	
ESP Subsystems	• Viewing looking aft unobstructed* • Thermal curtain design for capsule bus must be RF transparent*	• Antenna located with principal unit.	• Minimize Capsule Bus/ESP structural interface.
	• Jettisoning of de-orbit motor substructure to provide antenna viewing angle for ESP.	• Principal unit contains all ESP subsystems	
Other Trajectory/Atmosphere Reconstruction Aids	• High altitude altimeter* • Rate gyros*	• Extends capability to 200,000 ft • Monitor during entry for $(\theta-\gamma)$ time history	
	• De-Orbit monitoring*		

Figure 4.2–12

4.4 SURFACE LABORATORY SYSTEM (SLS) - The objective of the Surface Laboratory (SL) is to perform landed scientific investigations on Mars. A mission operations profile for the preferred concept, from prelaunch (on the pad) to landing on Mars, is shown in Figure 4.4-1. We have provided for continuous operation of the cruise commutator throughout interplanetary and Mars orbiting flight, in order to give a current status of the SL equipment, even when it is not active.

After landing, the active phase of the Surface Laboratory begins. The operation is completely autonomous and follows a preprogrammed series of events to establish its mission objectives. These events consist of activating the science support equipment, establishing Mars-to-Earth communications and the Earth-to-Mars command link, deploying experiment equipment and sensors, and performing the experiments.

The nominal mission is 28 hours. It is based on a morning terminator landing, which we prefer because of greater mission flexibility, better surface lighting conditions, and better landing visibility direct from Earth. However, the Surface Laboratory has the capability of being landed anywhere on Mars in the daylight within the latitude of 10° N to 40° S, and mission life can be extended above the nominal as much as 100%, depending on the conditions actually encountered.

All major engineering and science landing operations are conducted according to time references provided by the Sequencer and Timer (S&T). The major events of the 1973 landed sequence are shown in Figure 4.4-2.

The preferred Surface Laboratory is shown in Figure 4.4-3. The installation of the science support subsystems is highlighted in Figure 4.4-4. We have arranged these subsystems to facilitate accessibility and have utilized standardized modules wherever feasible. As noted, the batteries are installed as four identical units in a central location. The electronic equipment consists of standardized subassemblies which are modules of uniform width and height but of variable thickness. The support equipment can be located in any of four equipment racks within the geometry constraints of each particular item. This permits efficient grouping of interfacing equipment and provides a high degree of flexibility. The structure basically consists of support beams and rectangular structural sandwich panels.

Figure 4.4-5 highlights the installation of the science subsystem within the Surface Laboratory. All requirements for field-of-view or access to the surface have been met for the instruments chosen for the preferred design. The subsurface probe, the soil sample acquisition equipment, and the in situ life detectors are installed at the corners of Surface Laboratory to maximize their access to the Martian surface.

SURFACE LABORATORY OPERATIONS, EARTH LAUNCH TO MARS LANDING

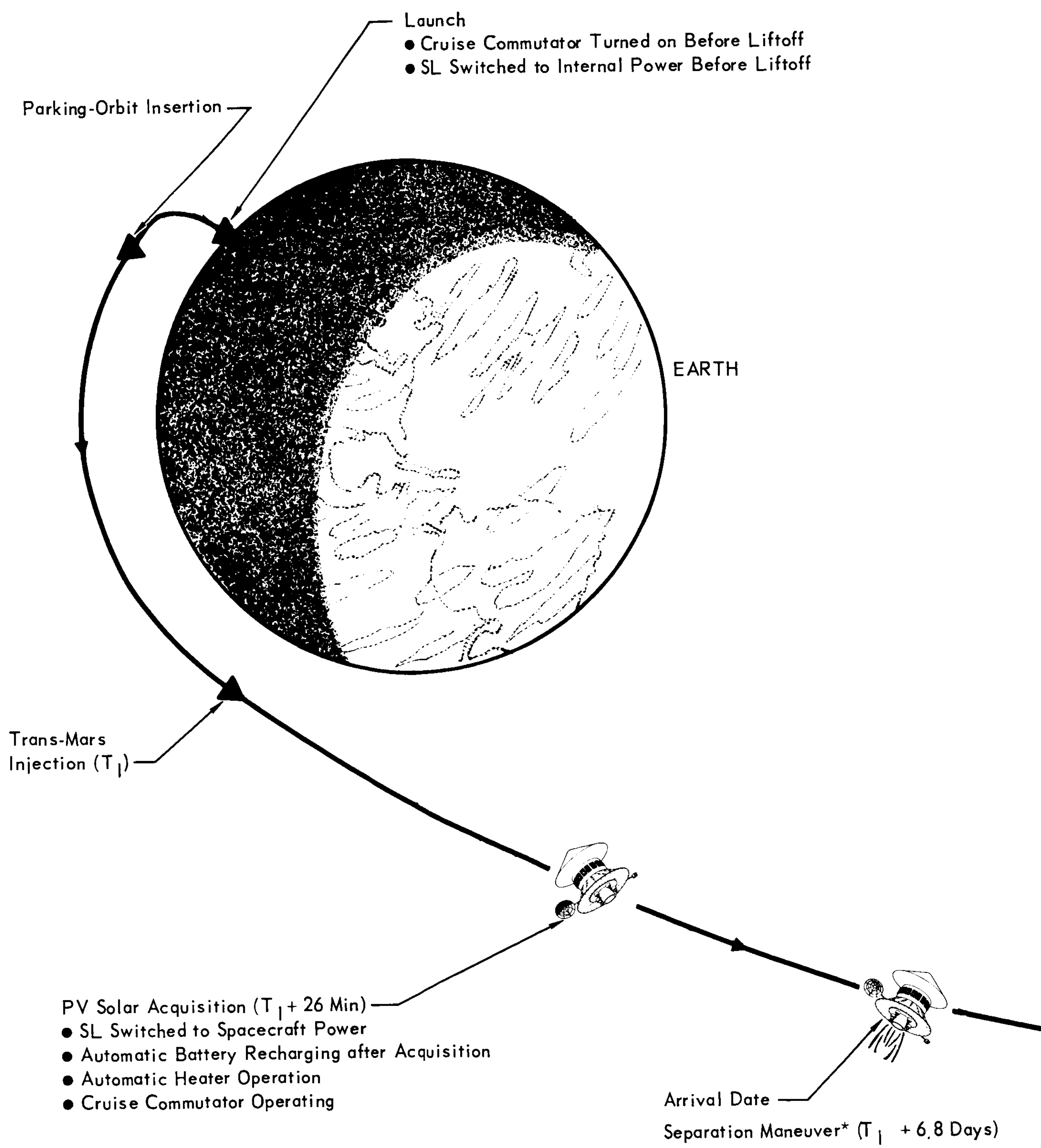


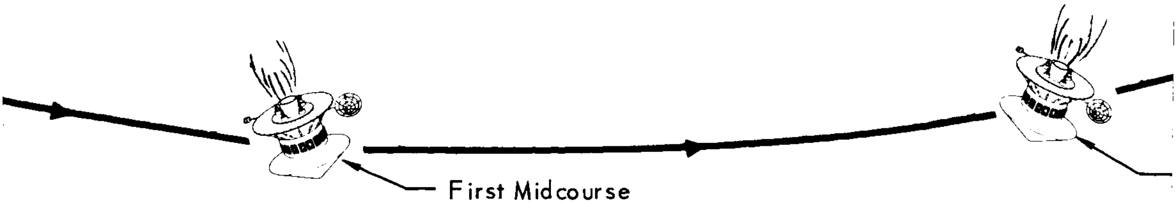
Figure 4.4-1

Spacecraft–Capsule Separation (T_{01}

- Subsystems Checked out 1 Orbit bef
- Cruise Commutator Operating as Pa
metry

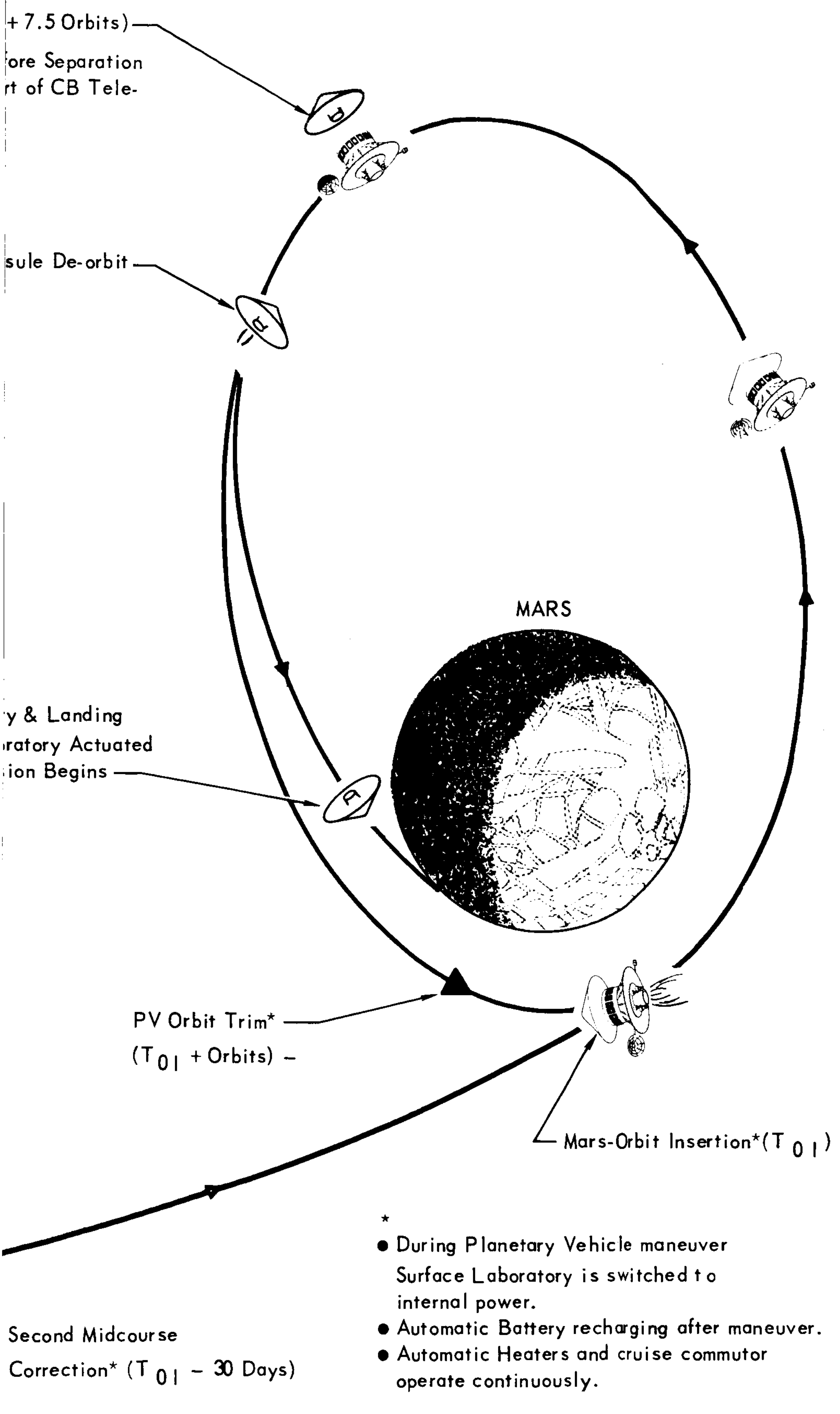
Cap

Capsule Entr
Surface Labo
Science Miss



First Midcourse
Correction* (T_{1} + 30 Days)

4-66-2



+ 7.5 Orbits) –
ore Separation
rt of CB Tele-
sule De-orbit
y & Landing
ratory Actuated
ion Begins
MARS
PV Orbit Trim*
(T 0 I + Orbits) –
Mars-Orbit Insertion*(T 0 I)
Second Midcourse
Correction* (T 0 I – 30 Days)
*
● During Planetary Vehicle maneuver Surface Laboratory is switched to internal power.
● Automatic Battery recharging after maneuver.
● Automatic Heaters and cruise commutor operate continuously.

TYPICAL MISSION SEQUENCE – 1973 OPPORTUNITY

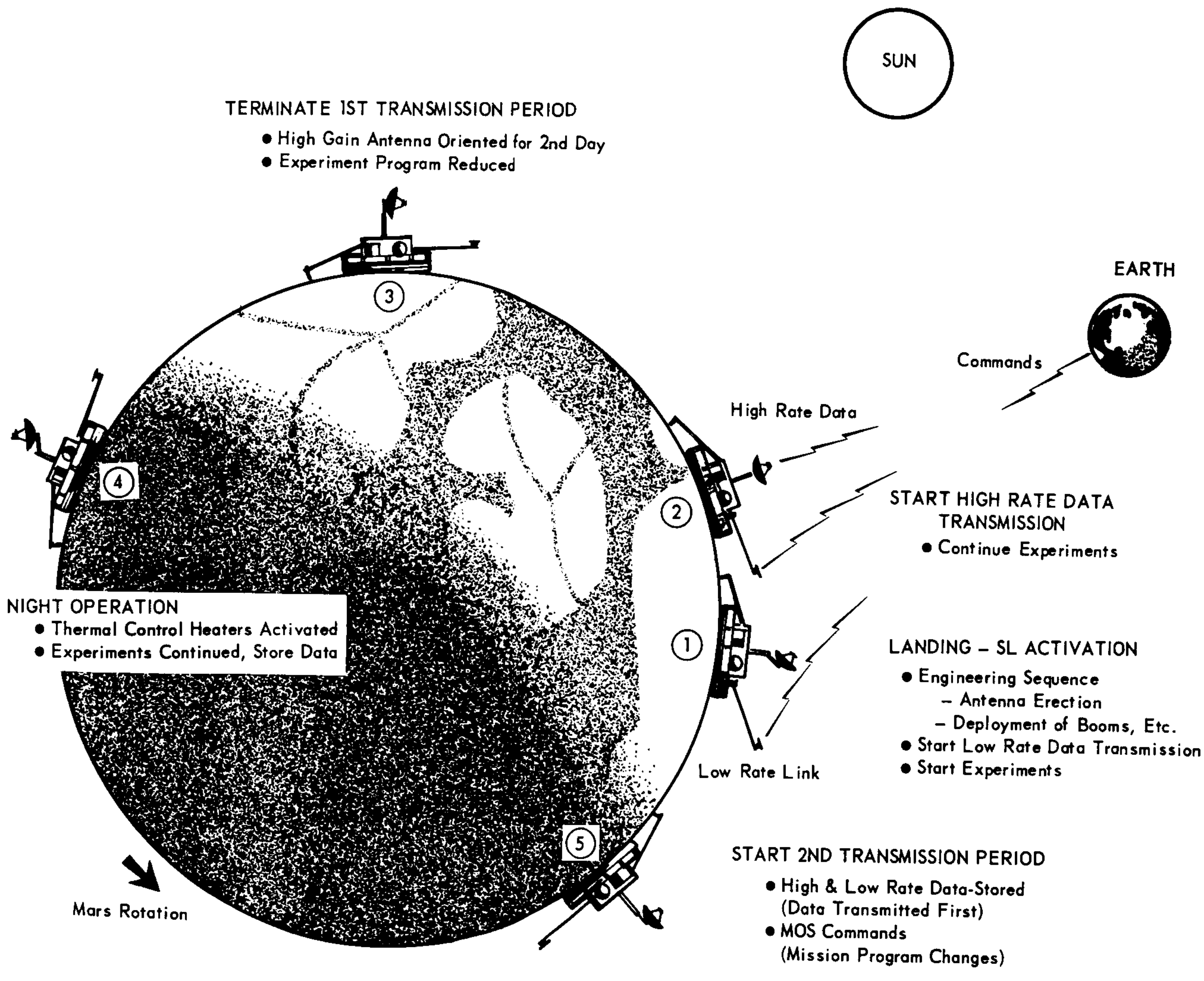


Figure 4.4-2

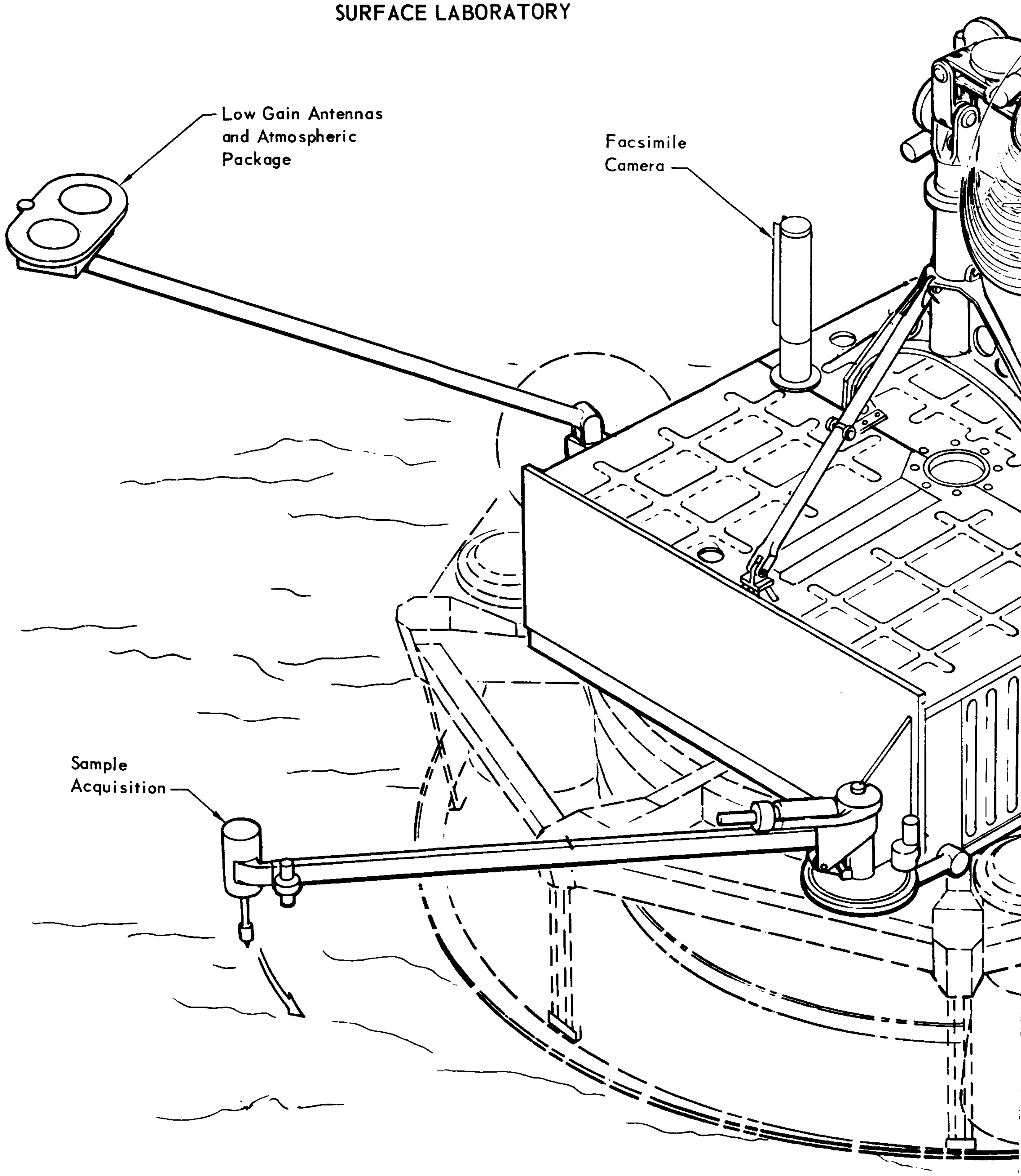
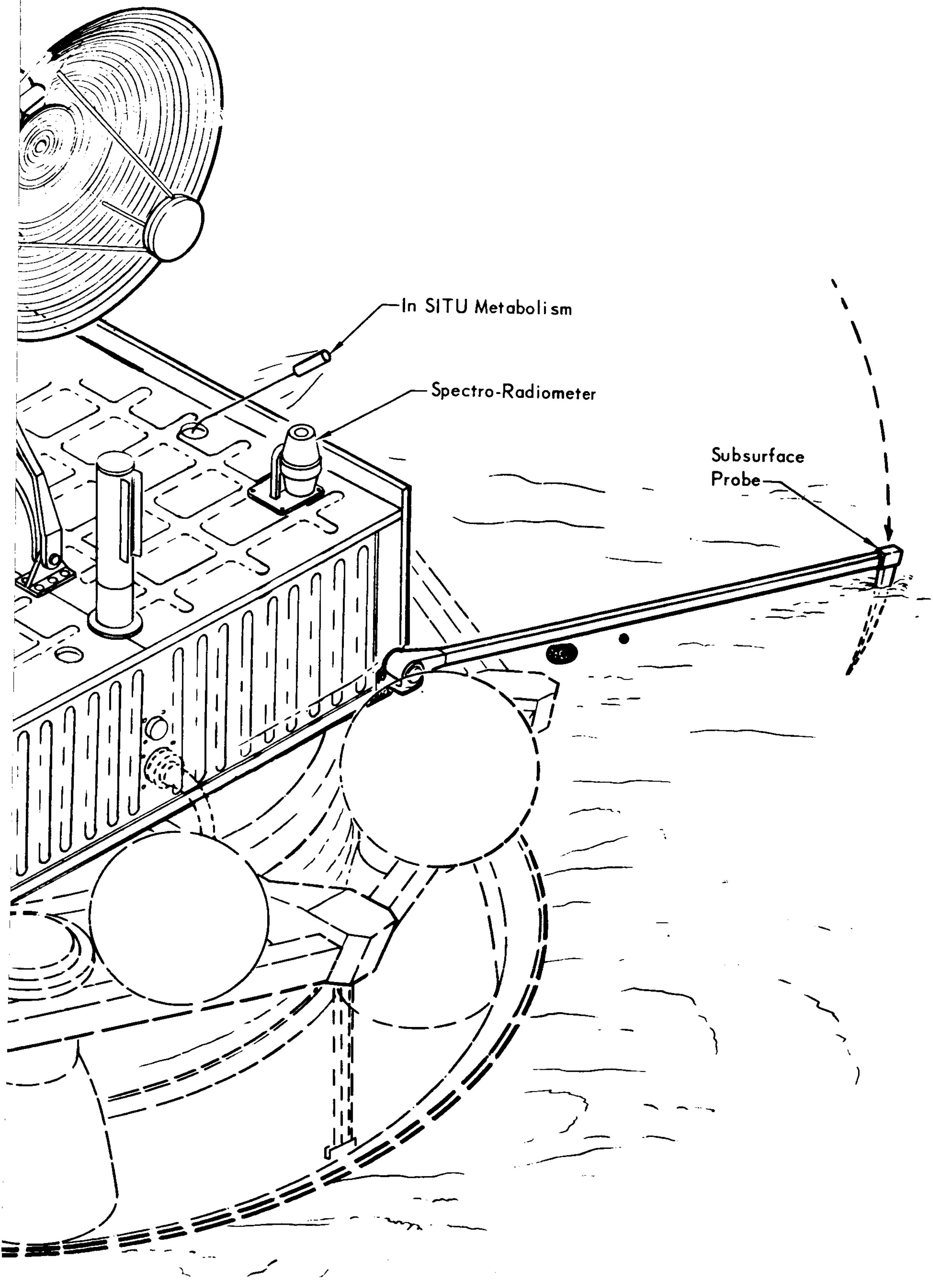

Low Gain Antennas
and Atmospheric
Package
Facsimile
Camera
Sample
Acquisition

Figure 4.4-3


In SITU Metabolism
Spectro-Radiometer
Subsurface
Probe

SUPPORT SUBSYSTEM EQUIPMENT LOCATION

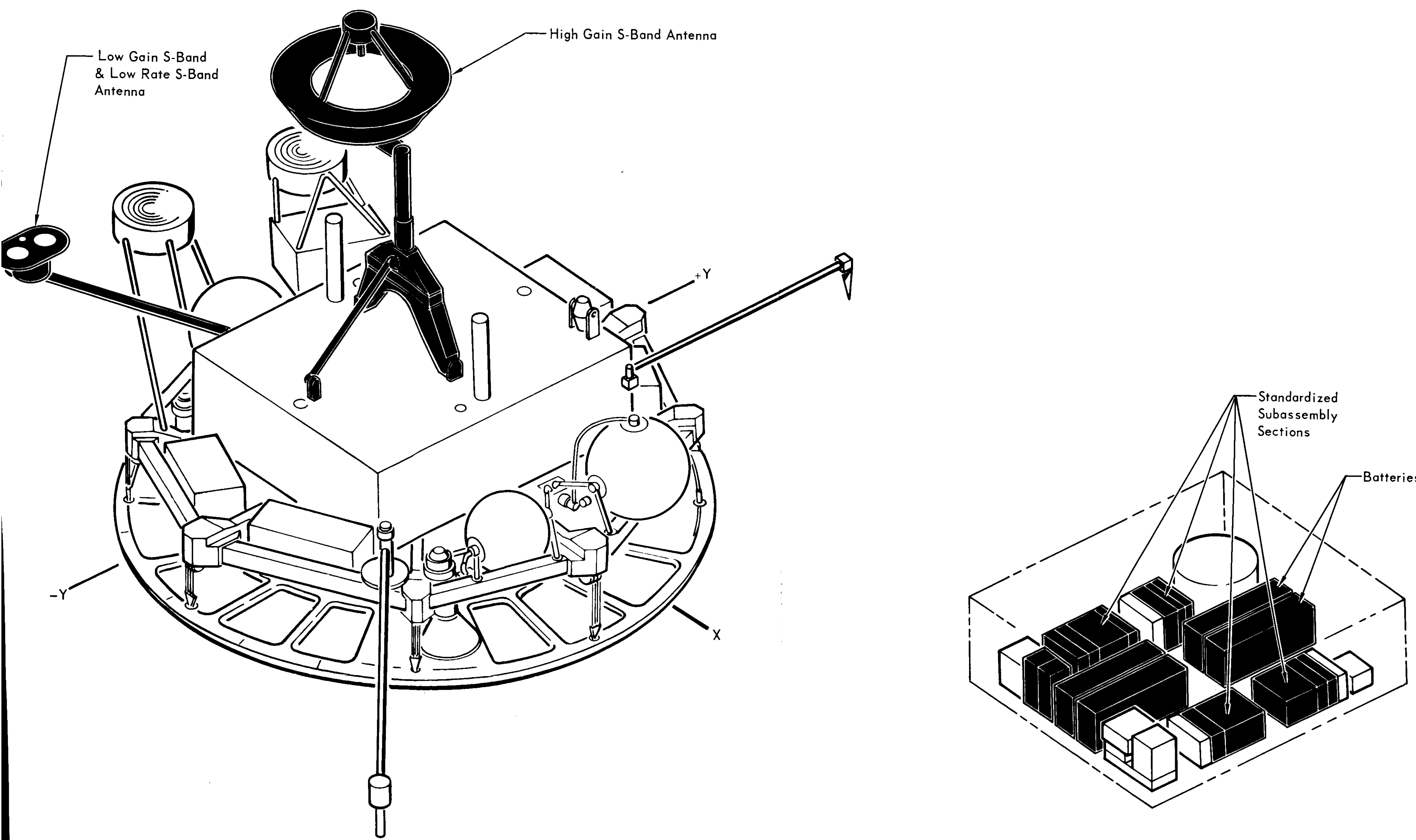


Figure 4.4-4

SCIENCE SUBSYSTEM EQUIPMENT LOCATION

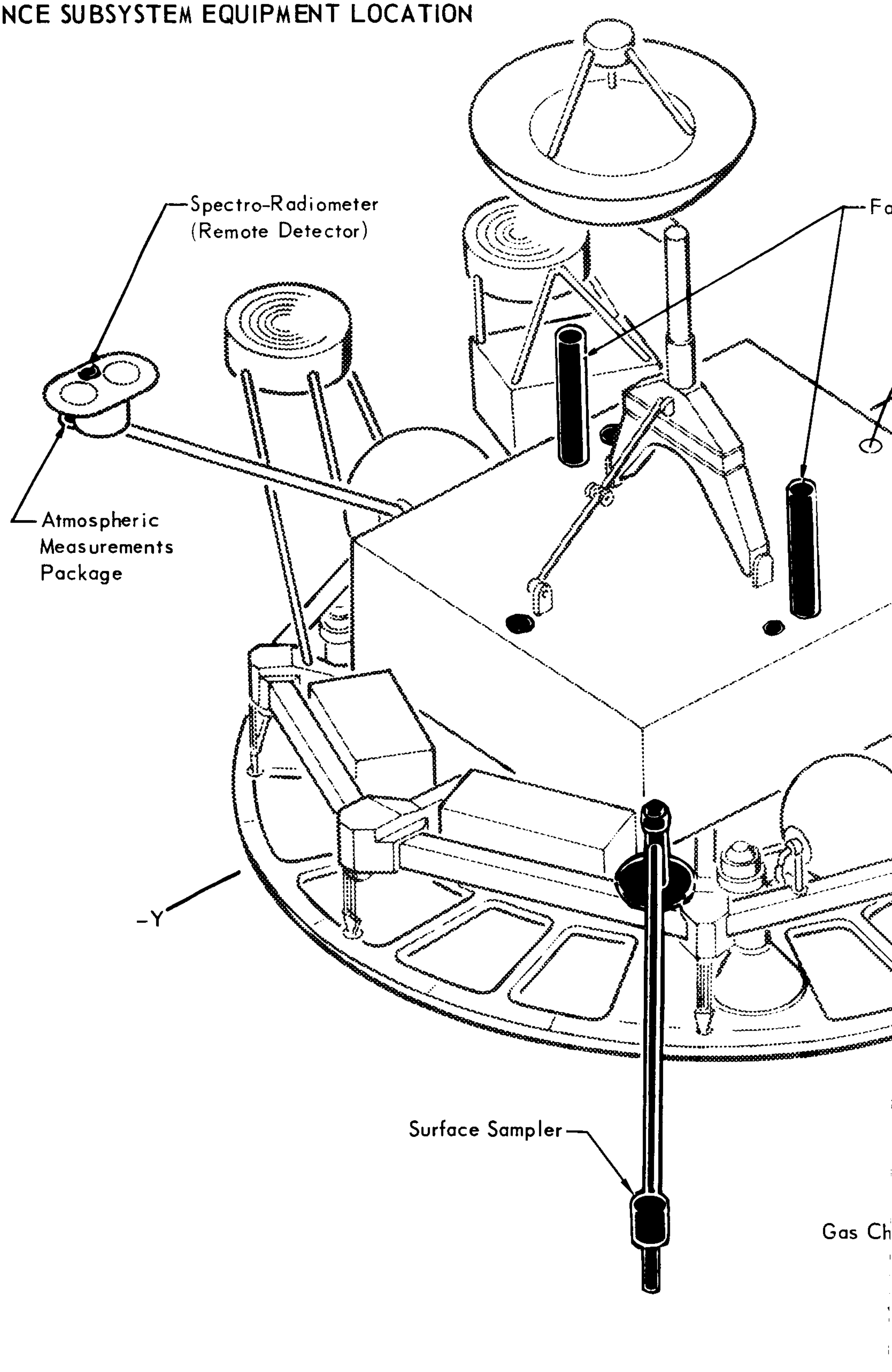
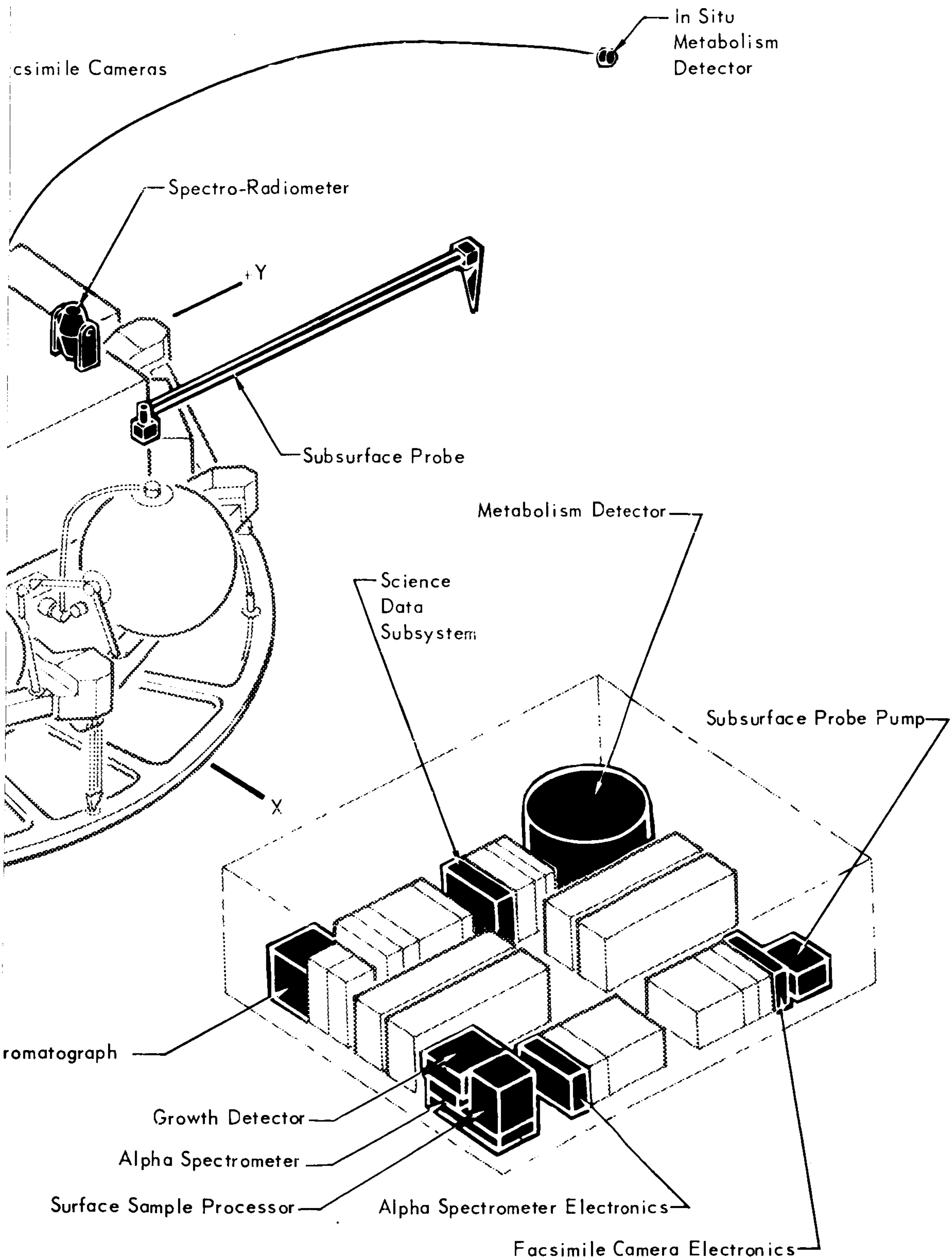


Figure 4.4-5


In Situ
Metabolism
Detector
csimile Cameras
Spectro-Radiometer
+Y
Subsurface Probe
Metabolism Detector
Science
Data
Subsystem
Subsurface Probe Pump
X
romatograph
Growth Detector
Alpha Spectrometer
Surface Sample Processor
Alpha Spectrometer Electronics
Facsimile Camera Electronics

The facsimile cameras, the spectro-radiometers, and the atomospheric package are extended beyond the normal confines of the SL to meet field-of-view requirements. In contrast to the Surface Laboratory support subsystems, the science instruments - aside from the data handling and instrument control equipment - do not lend themselves to standardized subassemblies, because they are tailored for specialized investigative purposes. Nevertheless - where possible - instruments that have a common need are grouped together. For example, the alpha spectrometer and the growth detector are located next to the sample processor, because of their need for soil samples.

The VOYAGER Capsule System Constraints and Requirements Document specifies that the Surface Laboratory weight (including the Entry Science Package) shall be at least 900 pounds for the 1973 opportunity. Our preferred design exceeds this minimum requirement by approximately 22%. This is shown in Figure 4.4-6, which presents a weight summary for the Surface Laboratory (reflecting the preferred design system requirements and design criteria). Empirically derived provisions for contingencies are included in the nominal properties to account for items not specifically considered in the estimates. A weight uncertainty of $\pm$ 94 lbs was calculated for the Surface Laboratory and the Entry Science Package combined, based on statistical variation and estimation techniques. As long as requirements and criteria are not changed, mass properties can be expected to fall within this tolerance. It is worthy of note that approximately 45% of the weight of the Surface Laboratory is attributed to the electrical power and telecommunication subsystems.

4.4.1 Science Subsystem - The science subsystem performs exobiological, biochemical, and planetological experiments on the surface of Mars for at least one diurnal cycle in 1974. These experiments include observations on the physics and chemistry of the Martian lithosphere and surface atmosphere. It is not expected that the experiments, and instruments, for the 1973 mission will be selected by NASA until about July 1968. However, for the purpose of designing the preferred Surface Laboratory, the typical description of the science subsystem in the JPL Constraints Document was used.

The preferred science subsystem, shown in Figure 4.4-7, consists of (1) a science data subsystem, (2) sample acquisition and processing equipment, and (3) science instruments. The total weight of the system is 130 lb, which includes equipment located both on the outside of the lab structure and inside the controlled thermal environment. (See Figure 4.4-8 for a view of the deployed equipment). There are no doors in the thermal insulation - only small ports for transporting the pre-

SURFACE LABORATORY
GROUP WEIGHT SUMMARY

	Before Aeroshell Separation	After Aeroshell Separation
Structure	93.1	93.1
Thermal Control	133.5	133.5
Tele-Communications	145.5	145.5
Sequencer, Timer and Test Programmer	16.0	16.0
Electrical Power	272.0	272.0
Experiments	110.0	110.0
Wiring and Mounting Provisions	145.4	145.4
Surface Laboratory Less E.S.P.	915.5	915.5
Entry Science Package	180.6	178.0
Total	1096.1	1093.5

Figure 4.4–6

PREFERRED SCIENCE SUBSYSTEM PERFORMANCE CHARACTERISTICS

ITEM	PERFORMANCE CHARACTERISTICS				
	WEIGHT Lbs	VOLUME in^3	POWER Watts	OPERATING Time hrs.	MAX ENERGY Watt-hrs.
Science Data Subsystem	(20)*	(400)			(591)
Data Processor	10	200	11.5	27.5	316
Science Sequencer	10	200	10.0	27.5	275
Sample Acquisition and Processing Equip.	(30)	(1630)			(90.5)
Surface Sample Acquisition Equip. (Boom)	16	1135	30	2	60
Surface Sample Processor	8	420	10	0.67	6.7
Subsurface Probe	6	220	2	11.9	23.8
Science Instruments	79.5	3550			510.8
Cameras	15	370	15	0.75	11.3
Atmospheric Sensor Package	4.5	74	6.7	5.6	37.5
Spectro Radiometer	5	80	2	0.7	1.4
Alpha Spectrometer	10	600	2	27	54
Gas Chromatograph	15	400	15	15.2	236
Life Detectors	30	2026	10.5	–	170
Subsurface Probe Sensors	–	–	0.1	6	0.6
Total	129.5	5580	–	–	1192.3

*Weight included in telemetry/subsystem

Figure 4.4-7

ES OF SURFACE LABORATORY DEPLOYED EQUIPMENT

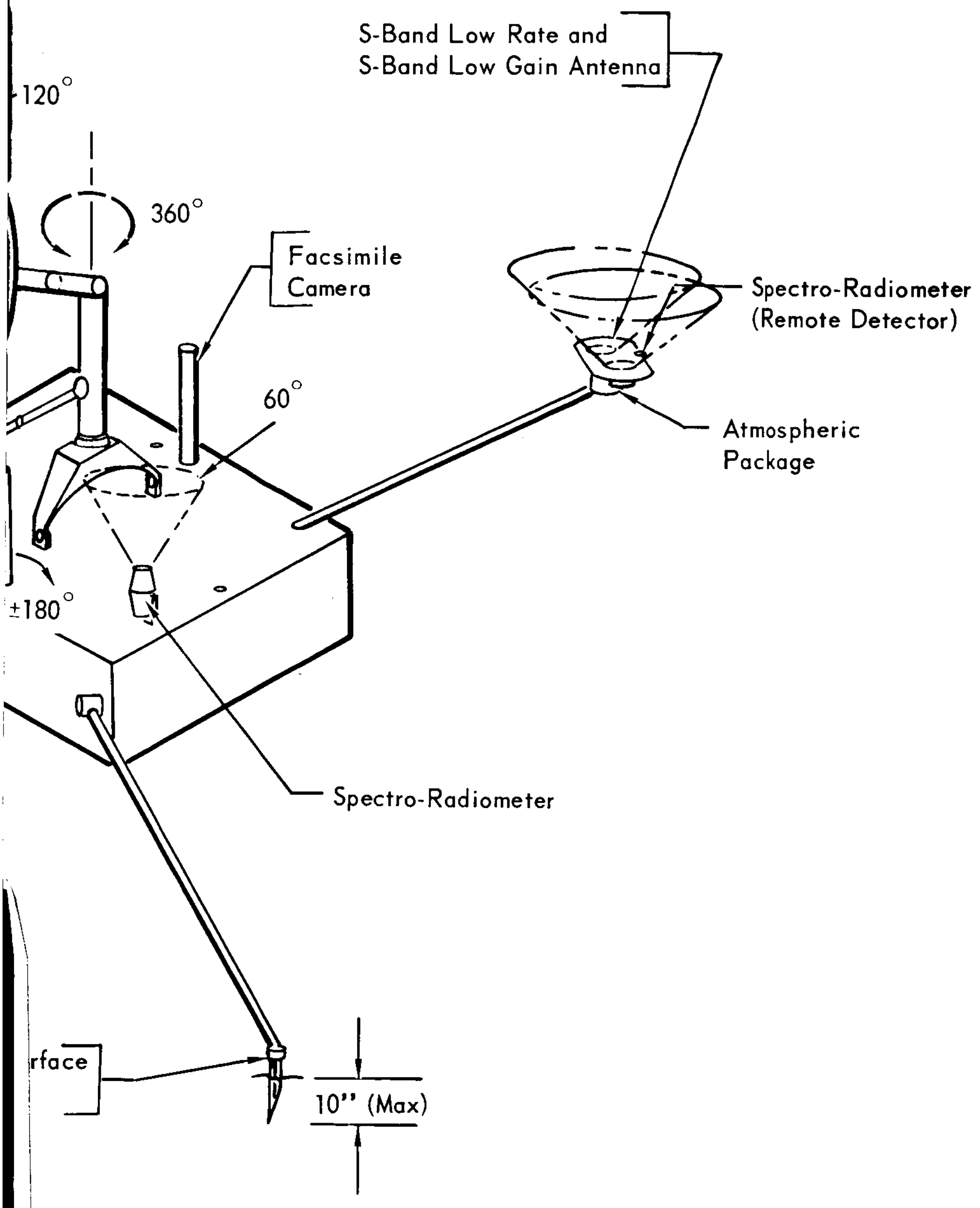


Figure 4.4-8

pared samples to the internal equipment. All mechanisms use a simple hinge type deployment. This design was selected to obtain simplicity for reliability.

For the preferred design, the major integration problems identified during our study are: (1) landing site surface contamination, (2) experiment thermal control, (3) experiment mechanical integration, and (4) electronic subsystem interfaces.

Surface contamination arises due to the interaction of the terminal descent propulsion engine plume with the landing site surface. This contamination could perturb the landing site environment enough that the scientific mission objectives could not be achieved. Most of the contamination interface problems are minimized by (1) terminating the Capsule Lander terminal descent engines 10 ft above the surface, (2) using a sampler to acquire samples to a depth of four inches, and (3) using remote in situ devices.

Experiment thermal control incompatibilities arise primarily because some of the experiment instrumentation is outside the thermally controlled laboratory interior. These problems have been resolved through judicious use of a combination of insulation and heaters.

The major mechanical integration problem is satisfying all of the view and access requirements of external experimental instruments and of other subsystems requiring external view (e.g., antennas). This problem has been resolved by comparing view angles in detail and by the use of folding booms where necessary.

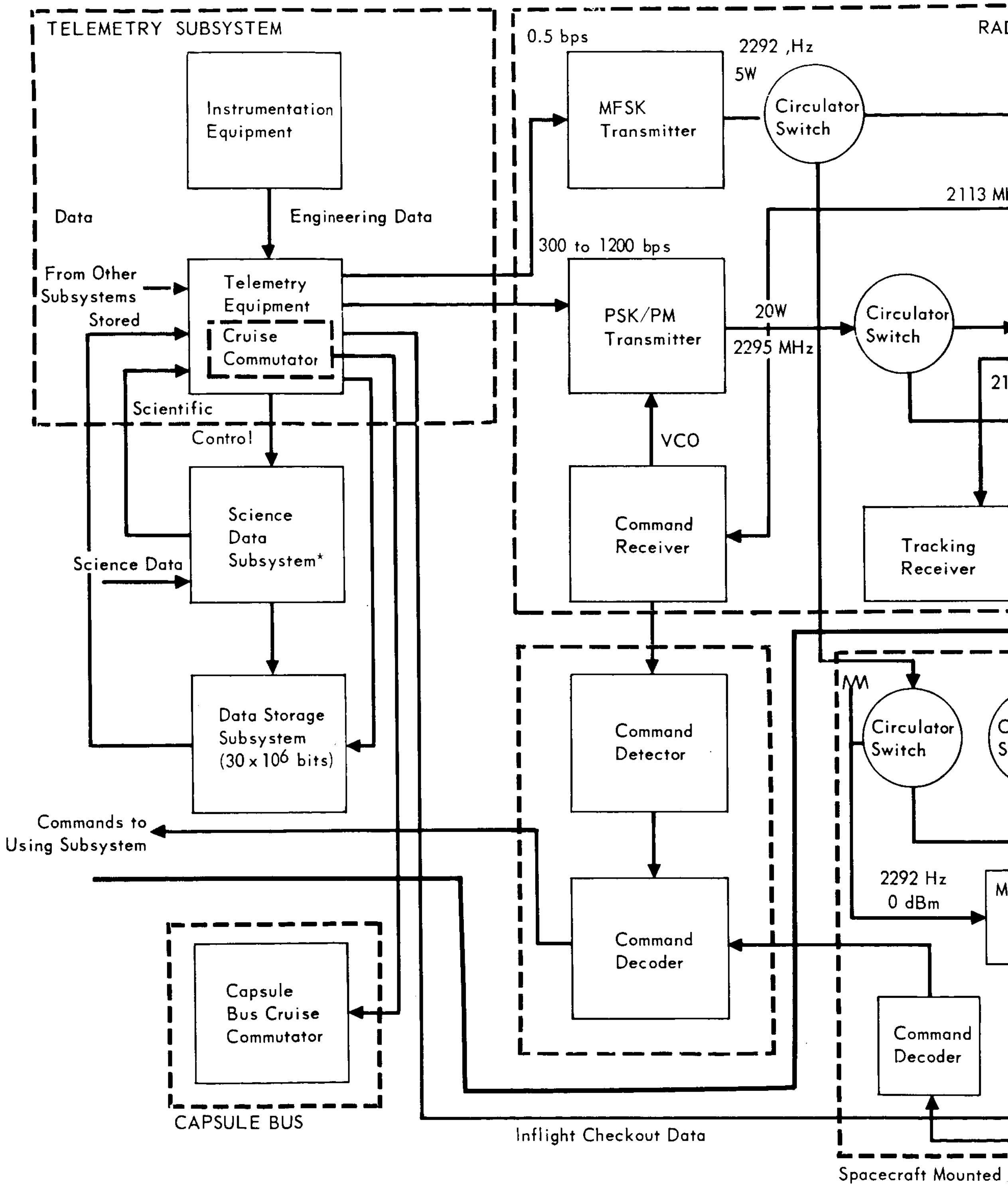
The requirement for interfacing experiment payloads that vary from mission to mission with supporting electronic subsystems which are standardized is the major electronics integration problem. This has been solved in our preferred science data subsystem design which provides remote interface units to satisfy the unique command and data conditioning requirements of individual experiments.

4.4.2 Telecommunications - The telecommunication functions for the SL consists of (1) reception of commands from Earth, (2) transmission of engineering and scientific data to Earth, and (3) metric tracking between the SL and Earth. A block diagram of the subsystem is shown in Figure 4.4-9.

The command function is performed by a 1 bps digital radio link similar to that used by the Mariner spacecraft. A low gain fixed-position receiving antenna with a 110° beamwidth is used for increased reliability and to maximize the length of time when the command link is available to Earth.

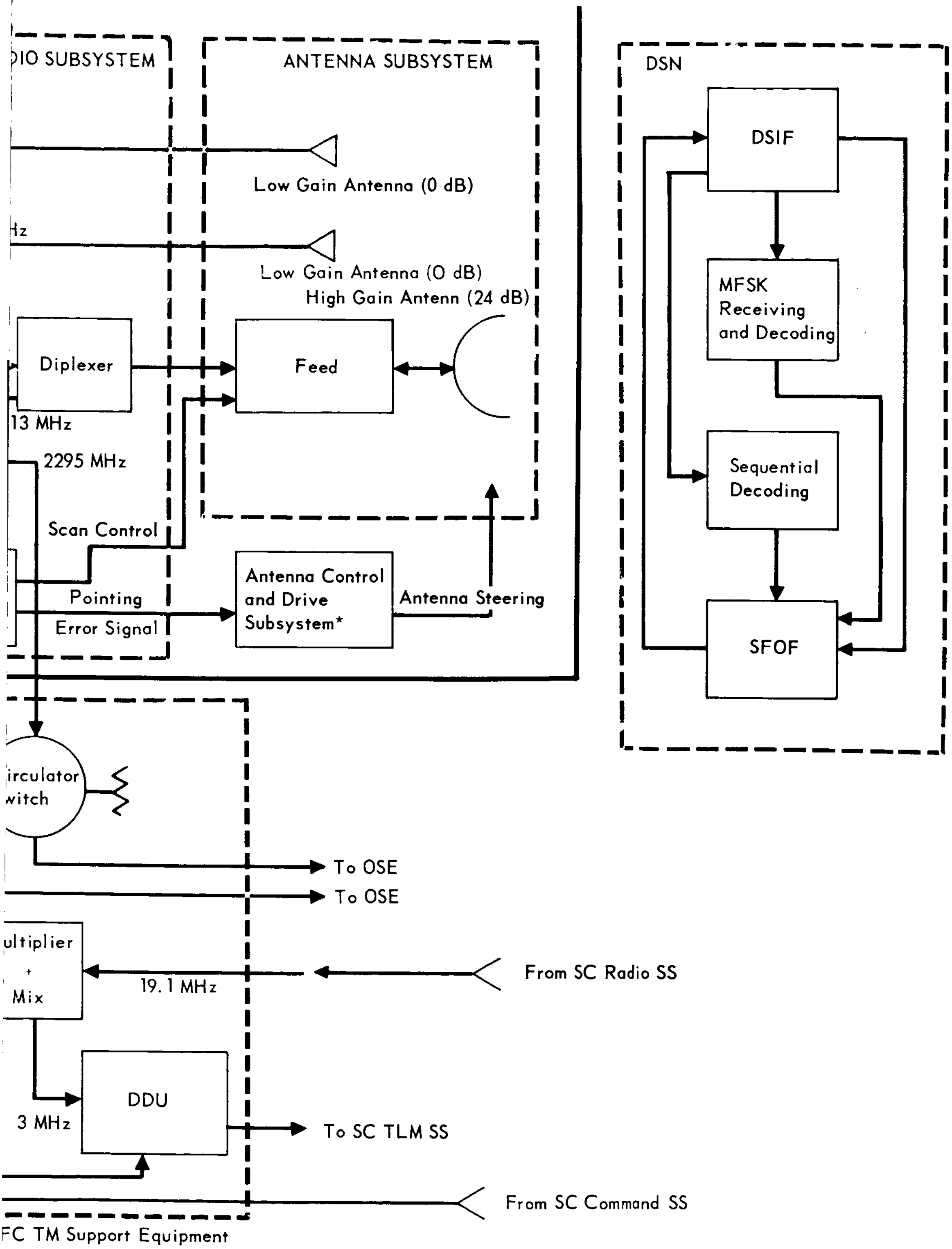
Essential engineering data are transmitted over a low rate radio link employing a 5 watt solid state transmitter and a separate low gain, wide beamwidth antenna.

SURFACE LABORATORY
TELECOMMUNICATIONS BLOCK DIAGRAM



*Not Part of Telecommunications

Figure 4.4-9



DIO SUBSYSTEM
ANTENNA SUBSYSTEM
DSN
Low Gain Antenna (0 dB)
Hz
Low Gain Antenna (0 dB)
High Gain Antenn (24 dB)
Diplexer
Feed
13 MHz
2295 MHz
Scan Control
Pointing
Error Signal
Antenna Control
and Drive
Subsystem*
Antenna Steering
DSIF
MFSK
Receiving
and Decoding
Sequential
Decoding
SFOF
irculator
witch
To OSE
To OSE
ultiplier
+
Mix
19.1 MHz
From SC Radio SS
DDU
3 MHz
To SC TLM SS
From SC Command SS
FC TM Support Equipment

4-75-2

Because of the low effective radiated power, a multiple frequency-shift keying (MFSK) modulation technique with a data transmission rate of 0.5 bps is employed.

Science data, plus additional engineering data, are transmitted over a third link. This link employs a phase-shift-keying/phase-modulation (PSK/PM) technique. Convolutional coding in the SL, with sequential decoding at the DSIF, is used to increase the data rate to 1200 bps maximum for the 1973 mission. A 36 inch parabolic antenna is oriented by a two-gyro system to align an hour axis parallel with the Mars axis of rotation. Antenna steering is provided by a clock signal. An auto-track back-up mode is provided, in which a tracking receiver and a switching four element feed arrangement generate pointing error signals from the carrier component of the received command signal.

Metric tracking consists of determining the relative velocity between the Earth DSIF and the SL by measuring the two-way Doppler shift on the command and high data rate carriers. The voltage-controlled oscillator in the command receiver controls the frequency and phase of the high rate transmitter when this measurement is made.

The telemetry equipment is controlled by a programmer that contains instructions in a core memory. These instructions can be changed by command from Earth, either prior to Flight Capsule separation or after landing.

Data not transmitted in real-time are stored in magnetic tape for later transmission.

The principal characteristics of the telecommunications subsystems are listed in Figure 4.4-10. The trades made in selecting the preferred design are summarized in Figure 4.4-11.

4.4.3 Power - We have selected a sterilizable silver-zinc battery power source, based on reasonable cost, system simplicity, small volume, low heat generation during peak power output, and absence of deleterious effects on the Martian environment. These considerations outweighed the weight penalty incurred in meeting the 8100 watt-hour energy requirements of the cloudy, cold day. Other candidates studied include:

a. RTG - Rejected for the short term mission due to excessive cost and excess weight.

b. RTG plus batteries - Rejected for the same reasons.

c. Fuel cell - Rejected at this time because of doubtful sterilization feasibility and the possible contamination of the Martian environment from purge gases, even though the potential weight saving is significant.

SURFACE LABORATORY
TELECOMMUNICATION PERFORMANCE CHARACTERISTICS

ITEM	PERFORMANCE CHARACTERISTICS					
	CARRIER FREQUENCY	DATA RATE	MODULATION	TRANSMITTER RF POWER	ANTENNA TYPE	SYNCHRONIZA
Command	2113 MHZ	1 bps	PSK/PM	10 kw into 210 ft. ant.	Cavity backed spiral	Psuedo Noise
Low Rate Data (Engineering)	2292 MHZ	0,5 bps	MFSK	5w	Cavity backed spiral	15 minutes cw tone followed by 2-tone syn signal
High Rate Data (Science)	2295 MHZ	300 600 or 1200 bps	PSK/PM	20w	36" steered parabola on 4-axis mount.	Derived from data

Figure 4.4–10

...TION	ACQUISITION TIME (DATA)	DATA MODES	TELEMETRY PROGRAMMER	REDUNDANT PATH ALTERNATIVES
	30 minutes maximum	--	--	Sequencer and timer provides control functions without up-date commands.
...c.	30 minute maximum	One: Post landing	Fixed format	Same data sent over high rate data link
	1 minute maximum	Four: • Inter-planetary deorbit/entry • Landed Day – Without imaging – With imaging • Landed night	Reprogrammable by command	• Antenna pointing by autotrack instead of by gyro/clock • Antenna pointing using 3 axis only • Minimum science data over low rate data link

SURFACE LABORATORY
TELECOMMUNICATION OPTIMIZATION STUDY SUMMARY

FUNCTION	CANDIDATES	EVALUATION FACTORS				
		PROBABILITY OF MISSION SUCCESS	SYSTEM PERFORMANCE	DEVELOPMENT RISK	VERSATILITY	SELECTION
Data Transmission	**Direct To Earth**	**Good**	**Good If Steered Antenna Used**	Similar	**Best – Independent Of SC Orbit**	x
	Relay Via SC	Poorer – Req SC To Operate	Good		Poor – Dependent On SC Orbit	
Transmitting Antenna Type	Fixed	Best	Low Capacity Cannot Meet Req For Data Transmission	Low	Poor – Restricted To Limited Sites And Times	
	Steered	**Less, But Adaquate By Redundant Design**	**Good – Can Exceed Req**	**Reasonable**	**Good**	x
Transmitting Antenna Design	Mechanically Pointed Array	Good	Good	Higher Than Parabola	Easier To Stow	
	Electronically Steered Array	Good	Good	Very High	–	
	Mechanically Pointed Parabola	**Good**	**Good**	**Low**	**Adaquate**	x
Data Link Coding	None	Good	Lowest Capacity	Lowest	–	
	Bi-Orthogonal	Good	2X Capacity	Low		
	Sequential Decoding	**Good**	**2.8X Capacity**	**Low**		x
Data Handling Configuration	Hardwire	Good – If No Late Changes In Experiments	Adaquate For Fixed Format	Low	Poor	
	Multiprocessor	Good	Good For All Conditions	High	Maximum Flexibility To Change	
	Hybrid	**Good**	**Good For All Conditions**	**Lower Than Multiprocessor**	**Good**	x
Low Data Rate Modulation	PSK/PM	Good	Low – Requires Power For Carrier Lock	Low	–	
	FSK W Bi-Orthogonal Coding	Good	Higher Capacity	High	–	
	MFSK	Good	Highest Capacity	Moderate		x

Figure 4.4–11

d. Solar cell plus battery - Rejected because of sensitivity to cloudy and cold day constraints.

e. Solar cell - Rejected for incompatibility with the wind, dust, and cloud cover design constraints.

The problem of battery cell degradation during the long interplanetary cruise has been overcome by providing continuous battery charging with a two-step float charge method that is based on the work performed by NASA Lewis Research Center in testing long-cycle silver-zinc batteries.

The Surface Laboratory electrical power subsystem provides electrical power redundancy for the Capsule Bus and the Entry Science Package, to improve reliability. The interconnection of these power subsystems requires a corresponding method of power return. Therefore, a single, common ground point is provided in the Surface Laboratory. This method provides the most reliability and least weight.

Figure 4.4-12 summarizes the power source trade study conducted for the electrical power subsystem and Figure 4.4-13 describes the subsystem characteristics of the preferred all-battery choice.

4.4.4 Thermal Control - The preferred method of thermally controlling the Surface Laboratory includes heat pipes, radiators, insulation, electrical heaters, and thermostats. The selected approach was screened initially from among nine candidate systems, which included various thermal control devices, such as thermal switches, louvers, movable insulation, etc. (See Figure 4.4-14). The optimization of the preferred system is described in Figure 4.4-15, which also shows the alternative element options that were examined. The alternate elements of the preferred concept were evaluated on the following criteria:

a. The total thermal control weight required for completion of the nominal morning terminator landing mission.

b. The capability for maintaining equipment temperature at acceptable levels.

c. The amount of extended mission capability.

d. The adaptability to off-design conditions of environment and equipment power levels.

The performance of the preferred concept is summarized in Figure 4.4-16. Although it was not the lightest weight combination analyzed, its adaptability to off-nominal design conditions and its ability to achieve the lowest maximum equipment temperatures during the daytime communication periods (100°F) were overriding factors.

SURFACE LABORATORY
ELECTRICAL POWER SOURCE TRADE STUDY SUMMARY

CANDIDATES	EVALUATION FACTORS					
	WEIGHT	RELIABILITY	COMPLEXITY	PERFORMANCE LIMITATIONS	COST	OTHER FACTORS
Battery	244 lb	.9895	4 Batteries 4 Chargers	8120 kWH (e) 9300 kWH (t)	$20,000	No contamination Sterilization feasible Minimum integration prob
Fuel Cell	122 lb	.9999	2 Fuel Cells 5 Tanks Plumbing	5500 kWH (e) 9300 kWH (t) 600 watts (e)	$400,000	Contamination possible Sterilization unknown Requires thermal integra
RTG	321 lb	.9981	4 RTG 2 Regulators 2 Converters	Unlimited Energy 300 watts (e) 6000 watts (t)	$28,000,000	Requires extensive therm integration Radiation contamination
RTG–Battery	342 lb	.9927	2 RTG 2 Regulators 2 Converters 2 Batteries 2 Chargers	Life limited by battery cycle life	$14,000,000	Requires thermal integra Radiation contamination Twofold development
Solar Cell–Battery	287 lb	<.9895	4 Batteries 4 Chargers 4 Regulators Solar Panel Isotope Heaters	Degradation in Martian atmosphere unknown	$250,000	Output depends upon ori of array Not compatible with win cloud cover constraint Volume not compatible w spacecraft

Figure 4.4–12

SURFACE LABORATORY
BATTERY SUBSYSTEM PERFORMANCE CHARACTERISTICS

ITEM	PERFORMANCE CHARACTERISTIC
Weight	244 lb.
Volume	3650 in^3
Available Electrical Energy	8120 Watt-Hours
Available Thermal Energy	9300 Watt-Hours
Peak Power Capability	Over 6000 Watts
Environment Contamination	None for Sealed Battery
Operating Life	18 Month Demonstrated 15 Cycles Demonstrated
Capacity Loss	Negligable on Float Charge
Operating Temperature	50°F to 120°F
Availability	Development Required
Sterilization	Feasibility Demonstrated
Reliability	.9895 – Nominal Mission Energy .9999 – Some Mission Energy
Components	4 Batteries 4 Battery Chargers

Figure 4.4–13

SURFACE LABORATORY THERMAL CONTROL CONCEPTS

INITIAL CANDIDATES

THERMAL CONTROL SYSTEM	DESCRIPTION	ADVANTAGES	DISADVANTAGES
1. Insulated Hinged Panels over Radiators	Panels opened for daytime heat rejection, closed at night – mechanically actuated.	Has best theoretical performance. Simple thermal design. Can provide sand abrasion protection for radiator thermal control coatings.	Possible interference with structure and experiemnts. Reliability of linkage actuators. Sand and dust in mechanism. Active – motor required.
2. Phase Change Heat Sinks	Controls temperature by alternately melting and freezing a self-contained material.	Passive, high reliability. Can be used to supplement other candidates.	Packaging and material development required. Heavy if used exclusively.
3. Mechanical Louvers	Movable louvers actuated by bi-metallic springs (or small motors).	Used previously on spacecraft.	Insufficient turn-down ratio to provide control in cyclic Mars environment. Reliability. Susceptible to sand and dust damage.
4. Thermal Switches and Radiator	Conduction controlled with bimetallic spring actuated contacts.	Used previously on Surveyor.	Relatively low heat transfer capability. Susceptible to sand and dust damage. Heat sterilization may damage actuator.
5. Heat Pipes and Radiator	Closed, self-contained system utilizing a capillary wick to return a heat transfer fluid from condenser (radiator) to evaporator (equipment cold plate). Controlled by pressure or temperature sensitive actuator which interrupts fluid flow.	Performance high Can adapt to late changes in equipment packaging and power. Controller internal, not affected by sand and dust, Mars atmosphere or heat sterilization. Element tests have shown feasibility.	New concept, not qualified as flight hardware. May require development of gravity independent control system and materials compatibility.
6. Liquid Water Evaporator	Self contained water storage boiler. Operation initiated by blowing pyro valve to vent water vapor to ambient.	Does not require radiators. Automatic operation.	Performance dependent on Mars atmospheric pressure. Vented water vapor may interfere with experiments. No extended mission capability.
7. Active Coolant Loop and Radiators	Uses radiator, coolant pump, cold plates, and associated plumbing. Thermostat control to actuate coolant pump.	Same as heat pipes. Insensitive to gravity. Used previously on Gemini.	Requires electrical power and development of low flow rate intermittent operating coolant pump. Low reliability
8. Passive System on Insulation and Heat Sinks	Depends on internal equipment and structure mass to absorb daytime equipment generated heat.	Simple Isolated from environment by insulation.	May be incompatible with equipment power and overheat. Sensitive to weight changes. Minimum adaptability to future missions.
9. Thermoelectric Devices (for Local Thermal Control)	Provides cooling and heating of equipment by thermoelectric principle. Suggested for life detection experiment temperature control only.	Only feasible means to achieve 0°C in Mars daytime environment. Proven concept (Gemini) High reliability.	Requires electrical power.

Note: (1) All systems considered may be used in combination with insulation and heaters for nighttime thermal control of primary equipment package or isolation of individual experiments.

(2) Heater candidates are electrical, radioisotope, and chemical.

Figure 4.4-14

SURFACE LABORATORY
THERMAL CONTROL SUBSYSTEM OPTIMIZATION STUDY SUMMARY

[redacted] Selected Candidate

FUNCTION	CANDIDATES	MERITS OF SELECTED APPROACH
Heaters	• Isotope [redacted] • Electrical and Isotope • Chemical	• Least complexity • Reliable • Controllable • No radiation effect on experiments
Number of Heat Pipes	• 2 heat pipes • 4 heat pipes • 8 heat pipes	• Best compromise for minimum weight and lowest maximum equipment temperature • Redundancy
Radiator Area (Total)	• Less than 15.5 ft^2 • Equal to 15.5 ft^2 • Greater than 15.5 ft^2	• Best compromise for minimum weight and lowest maximum equipment temperature • Minimum SL configuration interference
Radiator Position	• Horizontal [redacted] • Slanted	• Best performance • Minimum SL configuration interference • Minimum weight
Number of Radiators	• One • Two • More than two	• Best performance • Minimum weight • Redundancy
Thermal Control Coatings	• White porcelain enamel [redacted] • Ti O_2, Epoxy	• Low solar absorptivity • Simple to apply • Least susceptible to sand and dust erosion
Insulation and Heater Combination	• Maximum insulation, minimum heater power • Minimum insulation, maximum heater power [redacted]	• Minimum weight • Maximum extended mission with fixed weight
Minimum Equipment Temperature Design Environment	• Cyclic [redacted]	• Fulfills design constraint • Conservative insulation and heater battery weight • Maximum cyclic day mission extension
Maximum Equipment Temperature Design Environment	[redacted] • Continuous cloud cover	• Fulfills design constraint • Conservative heat pipe and radiator design • Conservative maximum equipment temperature
Nominal Mission	[redacted] • Evening terminator landing	• Lowest maximum equipment temperature • Maximum extended mission • Compatibility with mission environment and system constraints, i.e., communication time to Earth

Figure 4.4-15

SURFACE LABORATORY

THERMAL CONTROL SUBSYSTEM PERFORMANCE CHARACTERISTICS

ITEM	PERFORMANCE CHARACTERISTICS					
	EQUIPMENT TEMPERATURE -F°		NIGHTTIME PEAK HEATER POWER-WATTS		NIGHTTIME INTEGRATED HEATER POWER WATT HR	
	MAXIMUM - DAY	MINIMUM - NIGHT	CYCLIC	CLOUDY	CYCLIC	CLOUDY
Nominal morning landing	100	50	83	210	740	4000 (19 hrs)
Extended nominal mission - 1st day and night	77	50	100	**	1000	**
Evening terminator landing	115	50	83	210	740	6300 (30 hrs)

**Mission extension applicable only to cyclic normal mission.

NOTE – 116 watts of equipment disipated heat are not included.

Figure 4.4-16

4.4.5 Structure - The Surface Laboratory structure provides the support and mounting for the science instruments, the telecommunications equipment and power supply, and the thermal control subsystem. Its design is constrained by:

a. The thermal control devices required (heat pipes, insulation, radiators, and cold plate)
b. A minimum weight requirement of 900 lb for the total laboratory
c. Landing loads
d. The space provided by the Capsule Lander.

Figure 4.4-17 shows the structural arrangement for the Surface Laboratory. The design consists of support beams and a rectangular structural sandwich panel. The panel is continuous over the supports, provides for the support of the thermal radiators, and is connected to the radiators by heat pipes. The panel serves a dual purpose by carrying the equipment inertial loads to the framing members as well as providing a cold plate for the active equipment during daylight and heat retention for cold night operations.

Additional supporting structure, in the form of I-beams, trusses, and fittings, is provided for mounting the experiments and equipment.

Insulation is used as a thermal barrier for the Surface Laboratory and completely envelops the outside surface. It is 4 inches thick and is bonded to the metal panels. Ready access to equipment is provided by two removable insulation thermal covers on the upper surface.

4.4.6 Sequencer and Timer - The Sequencer and Timer (S&T) provides the Surface Laboratory with the means to accomplish, without primary Earth command, the automatic self-contained functions that are necessary for post-landing activation on the surface of Mars. Earth command resequencing is provided prior to Capsule Bus/Spacecraft separation and post landing.

The two major elements of the S&T in which alternate implementation techniques were considered are:

a. Memory storage technique (Preferred choice is magnetic core).
b. Timing technique (Preferred choice is the decrementing method).

The alternates studied and the major factors involved in choosing the selected techniques are presented in Figure 4.4-18. Performance characteristics of the preferred concept are given in Figure 4.4-19.

VOYAGER SURFACE LABORATORY – STRUCTURE AND INSULATION

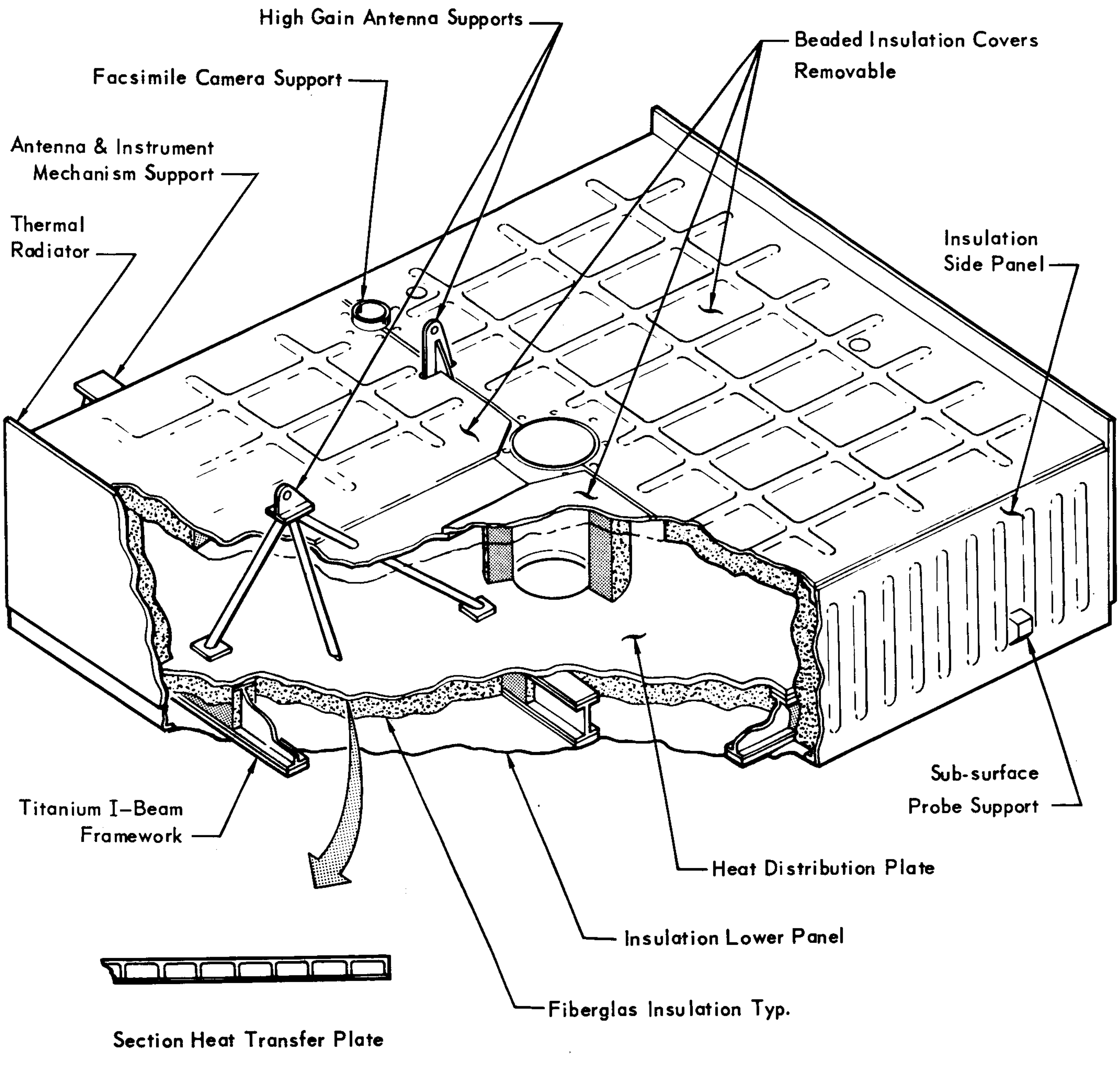


Figure 4.4–17

SURFACE LABORATORY
SEQUENCER AND TIMER OPTIMIZATION STUDY SUMMARY

Selected Candidate

FUNCTION	CANDIDATES	MERITS OF SELECTED APPROACH
Memory Storage Technique	[redacted] Magnetic tape/drum Semiconductor devices Advanced static magnetic devices	• Non-volatile memory • Less complex • More reliability, smaller, lighter and consumes less power • Survives Voyager environment/ sterilization • Lower development cost • Better development status
Timing Technique	Decrementing Method Incrementing Method	• Sufficient speed and accuracy • Less complex • More reliable • Easier design implementation • Best increased capacity ability • Greater flight experience

Figure 4.4–18

SURFACE LABORATORY
SEQUENCER AND TIMER PERFORMANCE CHARACTERISTICS

ITEM	PERFORMANCE CHARACTERISTICS
Weight	11 lb
Size	288 in.^{3}
Power Consumption	12 Watts (23 to 33 Vdc Primary Power)
Reference Frequency Outputs	1/240 Hz to 40 KHz, ± .01% Accuracy
Digital Word Output	16 Bits/Word at 500 to 40,000 bps
Discrete Command Output	32 Discretes, Delayed in Time – from any of 8 selected input occurrence.
Memory Size	128 Words – 24 Bits Each – Reprogrammable
Reliability	.9918 (Probability of 1973 Mission Success)

Figure 4.4–19

SECTION 5

CAPSULE GROWTH AND STANDARDIZATION

The VOYAGER mission goals will change throughout the 1970's as the environment becomes better understood and as the results of the early mission are evaluated. Since it is almost impossible to predict these changes, our preferred design incorporates a large degree of versatility. This versatility is a substantial advantage, both in assuring mission success during the 1973 opportunity - where operation in a new and unknown environment is required - and in enabling the use of the standardized systems in later opportunities, where advantage can be taken of the better understood environments.

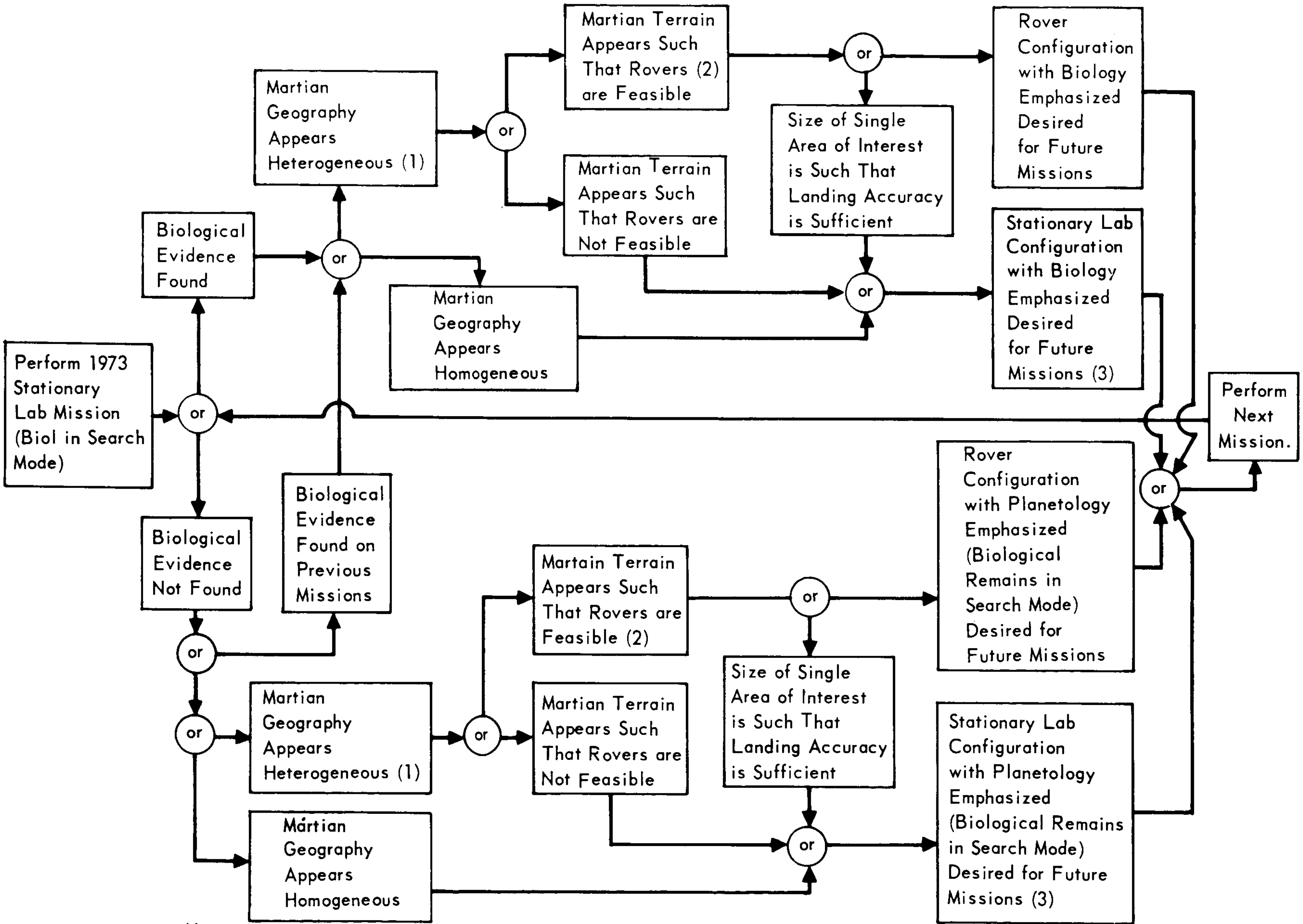
The key to selecting subsystems to be standardized - and exploiting the inherent advantages of standardization - is the balancing of the increasing performance requirements of later opportunities with the associated decreasing margin requirements for operation in environments that will be better known. The greatest opportunity for standardization exists in the Capsule Bus System. The use of standardized subsystems may degrade performance for individual missions, but the potential reduction in overall program cost and increase in reliability and operational flexibility are overriding.

5.1 GROWTH OBJECTIVES - The change in mission requirements for later opportunities is reflected in the evolutionary growth of the Surface Laboratory System. The definition of the 1977-79 Surface Laboratory System will depend strongly on the findings of the earlier missions. As shown in Figure 5-1, the candidates include both mobile laboratories and stationary ones (using small local rovers for sample gathering), emphasis being placed either on biology or planetology. In association, the preferred Capsule Bus System design provides growth to a gross weight of 7,000 pounds with a gross landed payload capability of about 1,900 pounds.

5.2 CAPSULE BUS STANDARDIZATION - As mentioned earlier, the two major influences on the Capsule Bus requirements for future missions are the increasing Surface Laboratory weight and the expected decreased in environment uncertainty. As observed, these are partially compensating. Our preferred design includes a substantial portion of standardization: approximately 85% by major assembly count, 78% by weight, and 80% by cost.

5.2.1 Operational Factors - The 1973 preferred design has a broad design envelope to accommodate the uncertainties of the Martian atmosphere and surface. As the early

POST 1973 CONFIGURATION ALTERNATIVES



Notes:

(1) Geography information provides basis for value of mobility and specifies its desired characteristics (sampling patterns, ranges, etc.)

(2) Terrain conditions coupled with desired payloads are such that minimum mobility requirements can be met.

(3) Stationary lab concept includes possible deployment of small local rovers for extended sample gathering. capability.

Figure 5–1

missions reduce the design uncertainties, the various subsystems can be operated closer to their design limits. For example, the Aeroshell has been designed for an M/C_DA of .3 slugs/ft^2 entering a VM-8 atmosphere at 15,000 ft/sec and a -20° entry angle - the critical conditions. Because of Surface Laboratory growth, the M/C_DA for the 1979 opportunity is .45 slugs/ft^2, which would impose an entry restriction with a lower entry angle or velocity (or both) if the actual conditions on Mars are still represented by the critical VM-8 atmospheric model. The VM-8 scale height of only 5 km is the dominant influence. If, however, the actual atmospheric scale height is determined from previous missions to be 7.2 km or greater, then the original 1973 design margin of 15,000 ft/sec and -20° entry angle can still be tolerated.

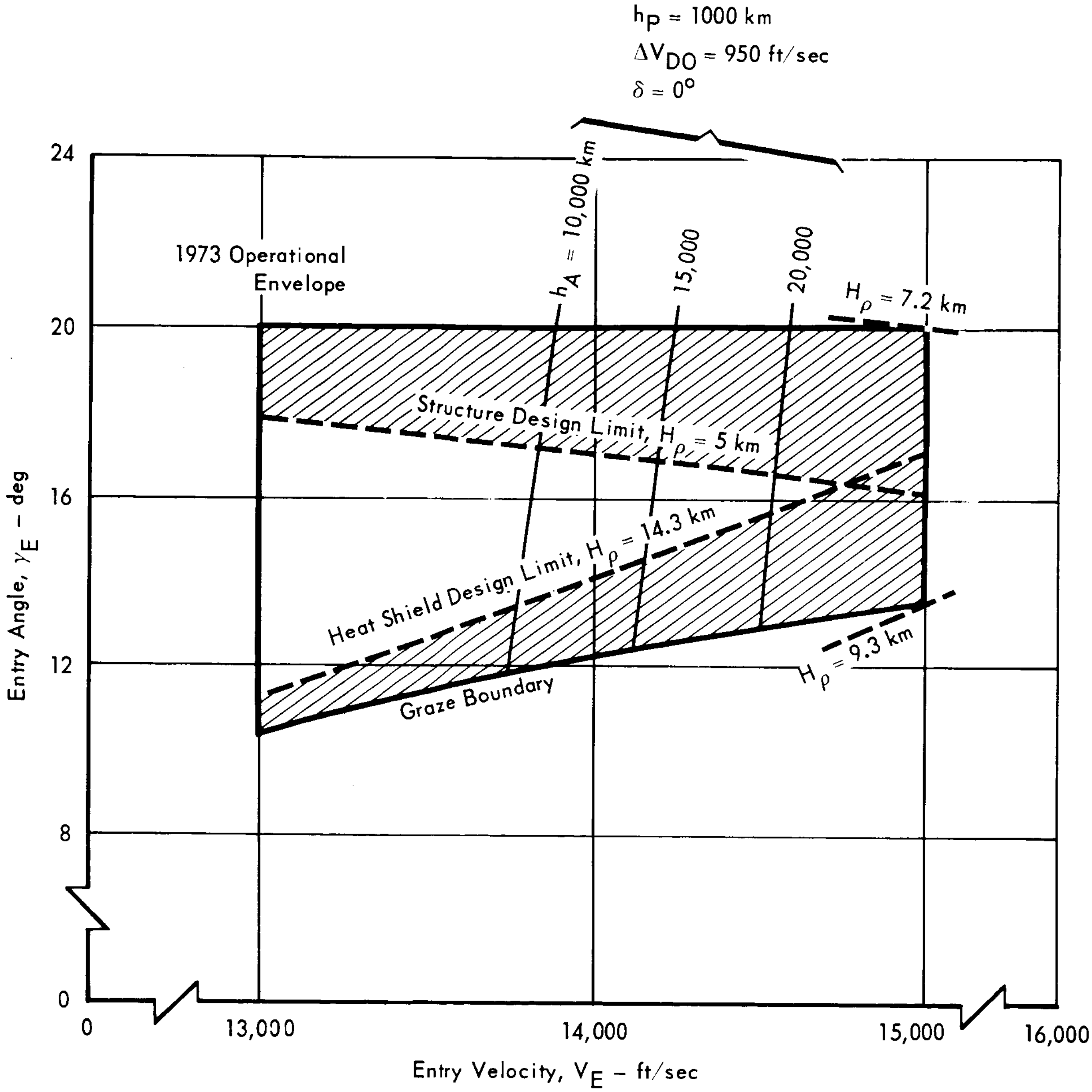
Figure 5-2 shows the moderate shrinkage in the entry velocity - entry angle (V_e - γ_e) envelope that results from the change in M/C_DA from .3 slugs/ft^2 in 1973 to .45 slugs/ft^2 in 1979, if the atmosphere scale height is found to have a spread between 5-14.3 km (VM-8 to VM-3). The entire envelope would be regained if the scale height uncertainty spread is reduced to the range of 7.2-9.3 km, with 9.3 km being the critical scale height for the heat shield design.

5.2.2 Subsystem Hardware Standardization - The sensitivity of the Capsule Bus subsystems to mission performance parameters is shown in Figure 5-3 and to environmental factors in Figure 5-4. The extensive standardization inherent in our preferred design is summarized in Figure 5-5. The major non-standard items occur in the structural elements; however, these tend not to be major development or long lead time items, and thus can be tailored for each opportunity. On the other hand, the expensive long lead time items have been highly standardized for post-1973 missions.

5.3 SURFACE LABORATORY GROWTH - The Surface Laboratory cannot be standardized in the same sense as the Capsule Bus. The planned growth from a 900 pound stationary laboratory operating for two days to a 1,900 pound, potentially fully mobile laboratory, operating for two years, as shown in Figure 5-6, effectively precludes a high degree of standardization. However, even here we have standardized the costly and long lead time subsystems wherever possible (See Figure 5-7).

OPERATIONAL ENVELOPE FOR 1979 MISSION

$M/C_DA = 0.45$ SLUGS/FT^2



h_P = Periapsis Altitude

h_A = Apoapsis Altitude

ΔV_{DO} = De-orbit Velocity Increment

H_ρ = Scale Height

γ = De-orbit Angle

Figure 5–2

CBS SUBSYSTEM SENSITIVITY TO MISSION PERFORMANCE

	LANDING WEIGHT	ENTRY WEIGHT	M/C_DA	POWER SOURCE	PAYLOAD TYPE	LANDING SITE
Aeroshell Structure			✓			
Heat Shield			✓			
Sterilization Canister				✓		
Adapter		✓				
De-orbit Propulsion		✓				✓
Terminal Propulsion	✓				✓	
Aerodynamic Decelerator	✓	✓	✓		✓	
Lander	✓				✓	
Reaction Control					✓	
Guidance & Control			✓			✓
Power				✓		
Telecommunications						✓
Thermal Control				✓	✓	✓

Figure 5–3

CBS SUBSYSTEM SENSITIVITY TO ENVIRONMENT DEFINITION

SUBSYSTEM	ATMOSPHERIC DATA	SURFACE DATA
Aeroshell Structure	Pressure vs. Altitude	
Heat Shield	Pressure vs. Altitude	
Canister		
Adapter		
De-orbit Propulsion		
Aerodynamic Decelerator	Density vs. Altitude	
Terminal Propulsion	Density vs. Altitude	Cohesiveness
Landing	Surface Winds, Density	Roughness, Slopes, Bearing Strength
Reaction Control	Low Altitude Winds	
Guidance and Control	Density vs. Altitude	Reflectivity, Roughness, Slopes
Power		
Telecommunications	Ionization Potential	Terrain Features
Thermal Control		

Figure 5–4

CAPSULE BUS STANDARDIZED HARDWARE LIST

FLIGHT CAPSULE WEIGHT 5000 LB IN 1973; GROWTH TO 7000 LB IN 1979

ITEM	STANDARDIZED				COMMENTS
	YES	PARTIAL	NO	DEGREE	
I. STRUCTURAL/MECHANICAL				Med	
1. Adapter		✓		Med	
a. Structural Assy.		✓			The general shape and structural concept
b. Canister Support		✓			but detail members will be beefed-up for th
c. Attach Fittings		✓			
2. Sterilization Assy.		✓		High	
a. Fwd Canister Assy.	✓				
b. Aft Canister Assy.		✓			Aft canister will have to provide for RTG missions.
c. Venting Assy.	✓				
3. Aeroshell		✓		Med	
a. Nose Cap Assy.	✓				RF transparency capability may influence
b. Heat Shield Assy.		✓			Ablative thickness may be changed to mee
c. Structural Assy.					
d. Radome & Window Assy.			✓		The window will not be required if the ES Flight Capsule.
e. De-Orbit Motor Support			✓		The heavier motor and different Surface L change the struts.
4. Lander		✓		Med	
a. Lower Equipment Assy.		✓			The configuration will not change but the beefed-up for increased loads.
b. Upper Equipment Assy.			✓		The configuration will be changed to mee Laboratory weights, shapes, and interface
c. Impact Assy.		✓	✓		Energy attenuator will be changed to mee
II THERMAL CONTROL				Med	The addition of RTG on later missions ha standardization of this system.
1. Heaters		✓		Med	Sizes may change to meet equipment requ
2. Thermostats	✓			100%	Number and location may vary.
3. Insulation			✓	Low	Insulation is tailored to the Surface Labo and mission requirements.
4. Coatings			✓	Low	Application is tailored to the Surface Lab and mission requirements.
III AERODECELERATOR				–	The degree of standardization is unknown
1. Aerodecelerator (Parachute)		✓			proves to be one of the denser models, th usable for the 1979 mission. However, if models, redesign will be required.
2. Structure and Mechanisms		✓		High	The design concept is standardized but w increased loads.
a. Deployment		✓			
b. Cover		✓			
IV ENTRY SCIENCE PACKAGE					Specialized equipment for the 1973 and p

Figure 5-5

ıre standardized
e increased loads.

ıeat transfer in later

a later change.
t mission requirements.

' is eliminated from the

ıboratory shape will

structure will be

different Surface
s.
mission requirements.

s greatest impact on

rements.

atory configuration

oratory configuration

. If the atmosphere
en this design will be
t should be the thinner

ill be beefed-up for the

ıssibly 1975 missions.

ITEM	STANDARDIZE		
	YES	PARTIAL	NO
V DE-ORBIT PROPULSION			
1. Spherical Solid		✓	
a. Rocket Motor	✓		
b. Nozzle with Ball Release	✓		
c. Igniter Assy.	✓		
VI TERMINAL PROPULSION			
1. Propellant Supply	✓		
a. Fuel & Oxidizer Tanks	✓		
b. Pyro Valves	✓		
c. Fill Valves	✓		
d. Filters	✓		
e. Check Valves	✓		
f. Burst Diaphram & Relief Valves.	✓		
2. Pressurant Assy.	✓		
a. Tank	✓		
b. Pyro Valve	✓		
c. Fill Valve	✓		
d. Filter	✓		
e. Regulator	✓		
f. Shut-off Valve	✓		
3. Throttable Engines		✓	
a. Throttling Valves	✓		
b. Shut-Off Valves	✓		
c. Access Ports & Plumbing			✓
VII REACTION CONTROL			
1. GN_2 Pressurant Assy.	✓		
a. Tank	✓		
b. Regulator	✓		
c. Pyro Valves	✓		
d. Fill Valves	✓		
e. Filters	✓		
f. Check Valves	✓		
g. Shut-Off Valve	✓		
2. Propellant Tank Assy.		✓	
a. N_2H_4 Tank	✓		
b. Fill Valve	✓		
c. Pyro Valve	✓		
d. Filters	✓		
e. Access Port & Plumbing			✓

D		COMMENTS
	DEGREE	
	High High	Inert ports standardized; propellant is off-loaded in 1973 by lowering the volumetric efficiency.
	High 100%	The tankage is sized for the 1973 mission to prevent excess weight penalty. Later missions (heavier vehicle) will require additional (one each) 1973 mission design tanks.
	100%	Same comment as above.
	High	May be changed to adapt to mission equipment mounting and installation changes.
	High 100%	
	High	Higher fuel usage for maneuvering but lower usage during cruise because the Capsule Bus inertia in 1979 balances or is better than usage rates for 1973.
✓		May be changed to adapt to mission equipment mounting and installation changes.

CAPSULE BUS STANDARDIZED HARDWARE LIST (Continued)

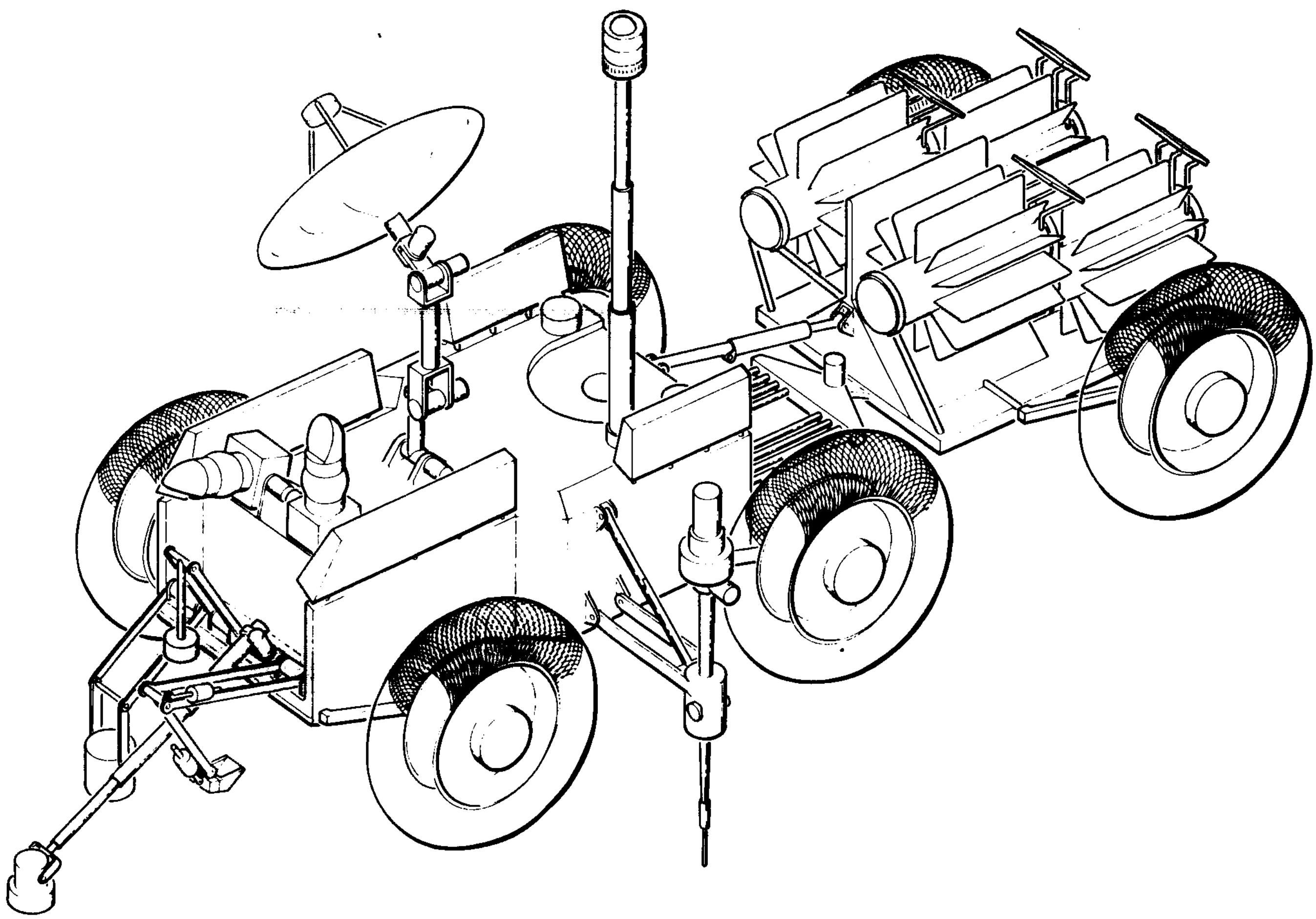
FLIGHT CAPSULE WEIGHT 5000 LB IN 1973; GROWTH TO 7000 LB IN 1979

ITEM	STANDARDIZED				COMMENTS
	YES	PARTIAL	NO	DEGREE	
3. Thrust Chamber Assy.	√			100%	
a. Thrust Chambers	√				
b. Propellant Valves	√				
VIII POWER				100%	
1. Bus Mounted Equipment	√			100%	
a. Battery	√				
b. Battery Charger	√				
c. Power Switching & Logic	√				May be programmed to meet mission requirements.
2. Adapter Mounted Equipment	√			100%	
a. Battery	√				
b. Battery Charger	√				
c. DC to DC Converter	√				
IX GUIDANCE & CONTROL				100%	
1. IMU & Support Electronics	√			100%	
2. Guidance & Control Computer	√			100%	Computer will be programmed to meet mission requirements.
3. Guidance & Control Power Supply	√			100%	
X SEQUENCER				100%	
1. Sequencer & Timmer	√			100%	Will be programmed to meet mission requirements.
2. Test Programmer	√			100%	
XI RADAR				100%	
1. Landing Radar	√			100%	
a. Antenna Assy.	√				
b. Electronics Assy.	√				

Figure 5-5 (Continued)

ITEM	STANDARDIZED				COMMENTS
	YES	PARTIAL	NO	DEGREE	
2. Radar Altimeter	√			100%	
a. Electronics	√				
b. Altimeter Antenna	√				
XII TELECOMMUNICATIONS				100%	
1. UHF	√			100%	Minor items removed on later missions for Entry Science Package elimination.
a. UHF Diplexer	√				
b. Transmitters	√				
c. Cruise Commutator	√				
d. DAS	√				
e. Parasitic Antenna	√				
f. Antennas	√				
2. Instrumentation	√			100%	
a. Pressure Transducers	√				
b. Temperature Transducers	√				
c. Acceleration Transducers	√				
d. Analog Digital Converter	√				
3. Spacecraft Mounted Equip.	√			100%	
a. RF Receivers	√				
b. Antenna	√				
c. Data Handling	√				
c. Data Handling	√				
XIII PYROTECHNICS				High	
1. Release Mechanisms		√		High	Some devices may be redesigned for later missions.
2. Initators (EED)	√			100%	The EED and circuitry are standardized for the Flight Capsule.
3. Circuitry	√			100%	

MOBILE LABORATORY CONFIGURATION



MOBILE LABORATORY WEIGHT STATEMENT	WEIGHT (lb)
Analytical Instruments and Detectors	198
Sample Collection and Processing	38
Telecommunications	148
Power (RTG's, Battery, Shielding)	390
Thermal Control (Insulation, Heat Pipes)	140
SLS Structure	100
Mobility System	600
Mobility Guidance	30
Wiring, Supports, Misc	150
Contingency	96
	1890

Figure 5–6

SURFACE LABORATORY STANDARDIZED HARDWARE LIST
FLIGHT CAPSULE WEIGHT 5000 LB IN 1973, GROWTH TO 7000 LB IN 1979

ITEM	STANDARDIZED			DEGREE	COMMENTS
	YES	PARTIAL	NO		
1. Power					
a) Batteries			√	Low	a) Ag/Zn batteries are used on the 1973 mission for their high energy density. On later missions, with RTG electric energy sources, the number of charge and discharge cycles precludes their use.
b) Power Control	√				b) Inverters, switching control, and battery charging control may be added in blocks to handle increased power requirements.
2. Sequencer & Timer	√			High	Excess capacity used for redundancy in 1973 used for higher capacity in later missions.
3. Command	√			High	Excess word decoding capability furnished.
4. Telemetry		√		High	Reprogrammable core memory
5. Data Storage		√		High	Designed in block data units which may be added for larger capacity. However, additional data control is required for each block added.
6. Radio		√		Med	Using reduced data rate for post 1973 missions could standardize transmitters or use of multiple transmitters is possible.
a) High Rate Link Power Amplifier					
b) High Rate Link Modulator	√				
c) Low Rate Link Power Amplifier		√			Used for initial mission operations, which will be at comparable ranges for future missions, If greater range required, need higher power transmitters.
d) Low Rate Link Modulation	√				
e) Command Receiver			√		Receiver sensitivity maximized for 1973 transmission distances.
7. Antenna		√			The high gain antenna may be used if RF power is increased or data rate is lowered.
a) High Gain Antenna Element					
b) High Gain Antenna Pointing Control		√			The gyrocompassing method of orientation suffers a degradation in accuracy as landings are made at latitudes greater than 40 degrees.
c) High Gain Antenna Diplexer etc.	√				
d) Low Gain Antenna Element		√			This is valid only if DSN Effective Radiated Power is increased.
e) Low Gain Antenna Diplexer, Etc.	√				
8. Science Data			√		Each interface and control unit is unique to the science instrument.
9. Thermal Control				Med	Heat pipes are relatively insensitive to changes in equipment heat dissipation.
a) Heat Pipes	√				
b) Heaters		√			Some heaters would be eliminated with RTG's.
c) Radiators	√				
d) Insulation		√			Reduced with RTG's
e) Coatings		√			
10. Structures		√		Low	Flat pallet design permits minimum modification to incorporate future mission science experiments.
11. Pyrotechnics		√		High	
a) Release Mechanisms					Some devices may be redesigned for later missions.
b) Initiators (EED)	√				
c) Circuitry	√				

Figure 5-7

SECTION 6

PLANETARY QUARANTINE

Planetary Quarantine requires that a probability of 10^{-3} of any one vehicle landing one viable earth organism on Mars shall not be exceeded. This constraint imposes a requirement for capsule sterilization, with attendant design implications.

We find that the key to efficient sterilization is to design a vehicle which can be effectively sterilized throughout, at 125°C for 24.5 hours. The problem here is not associated with equipment sterilizability, but in providing heat paths to all areas of the capsule in order to expose them for the prescribed temperatures and times. For example, insulated compartments, which are designed to provide equipment thermal control during operation on Mars, inhibit the attainment of internal temperatures during sterilization. In this case, our solution incorporates special heaters, which have been sized for sterilization requirements only, to supplement the external heating provided by the terminal heating facility. Provisions have also been made for sterilization of the interior of hermetically sealed assemblies during the Flight Acceptance heating cycle (which exceeds the terminal heating cycle) before installation of these assemblies into the vehicle.

6.1 MAJOR PLANETARY QUARANTINE REQUIREMENTS - Figure 6-1 lists the major planetary quarantine constraints specified by the VOYAGER Capsule System Constraints Document and our approach to complying with these requirements.

6.2 STERILIZATION COMPATIBILITY TESTING - The compatibility of all elements of the Flight Capsule with the temperatures required for terminal sterilization is verified during two test phases: qualification testing and flight acceptance testing. Qualification sterilization is specified at all assembly levels for which qualification testing will be performed. Flight acceptance sterilization testing is performed on all flight hardware to limits which are more stringent than the terminal heat sterilization cycle. Figure 6-2 shows a functional flow of the sterilization and testing interface.

MAJOR PLANETARY QUARANTINE REQUIREMENTS

REQUIREMENT Source: 1973 Capsule Systems Constraints and Requirements Document Revision 2	COMPLIANCE
1. Canister and Adapter for sterilization control After separation from Capsule Bus, the trajectory of separated canister shall not violate the planetary quarantine constraint.	Sterilization Canister A Sterilization Canister is designed which encapsulates the Flight Capsule during the period from before terminal sterilization to capsule separation in the Mars orbit. The canister provides a sterile dry nitrogen atmosphere at positive differential pressure throughout post-sterilization and prelaunch operations and automatically programs pressure relief during launch. A parting line seal maintains biological integrity. Attitude and separation velocity are controlled to prevent trajectories which violate the planetary quarantine constraint.
2. Sterilization Flight Capsule equipment designed to enter the Martian atmosphere shall be heat sterilized such that the probability that a live organism will survive the sterilization is less than 10^{-3}. The terminal sterilization cycle shall be consistent with less than 1×10^5 viable spores remaining during vehicle assembly. The temperature shall not be more severe than 125°C for 24.5 hours, as applied to the coldest contaminated point. In-flight sterilization of capsule hardware shall not be considered.	Sterilization Plan Flight equipment is designed to be compatible with the terminal sterilization temperature and time requirements and is composed of type approval qualified hardware. During manufacture, assembly, and testing, contamination controls are employed to assure that the Flight Capsule has fewer than 1×10^5 viable spores before entry into the terminal sterilization process. Within the sterilization oven, the Flight Capsule, which has been encapsulated in the Sterilization Canister, is heated at 125° C in a dry nitrogen atmosphere for a sufficient time to kill the accumulated spores. Achieving the required probability of survival of one living organism will require less than 24.5 hours. The sterility of the Flight Capsule is maintained by the Sterilization Canister during post-sterilization system tests, the mating with the spacecraft, launch, and Mars transit.
3. Sterilization of Interiors During Flight Acceptance Testing The interiors of certain specified items may not be required to reach sterilization temperature during the terminal sterilization cycle; however, during Flight Acceptance Testing the interiors of all Flight Capsule items must be subjected to an approved time/temperature cycle.	Sterilization of Insulated and Deeply Encapsuled Items Some equipment and subsystems require active thermal control and efficient insulation for Mars operation. Once such insulation is installed, it makes terminal sterilization heat soak periods become prohibitively long. In these cases, the following procedure is employed: • The interior of assemblies that are biologically (hermetically) sealed are sterilized during the Flight Acceptance heating cycle (which exceeds the terminal heating cycle) prior to their installation into the vehicle. This assures internal sterility. • Assemblies or subsystems enclosed within heavily insulated compartments (i.e., the Surface Laboratory) are heated to terminal sterilization temperatures by special heaters which supplement the external heating facility.
4. Acceptance of Flight Capsule Sterility Flight Capsules that have been subjected to an approved terminal sterilization cycle must be certified to have met the required probability of sterilization.	Certification of Flight Capsule Sterility The McDonnell Planetary Quarantine Manager certifies that each Flight Capsule is sterile after: • The number of viable spores on each Capsule is shown to be less than 10^5, as documented by reports from the Contamination Data System after the proper terminal sterilization parameters are imposed (and documented) by a Sterilization Engineer and verified by Quality Assurance inspection. • Sterility procedures are verified by complete biological examination of engineering models.

Figure 6–1

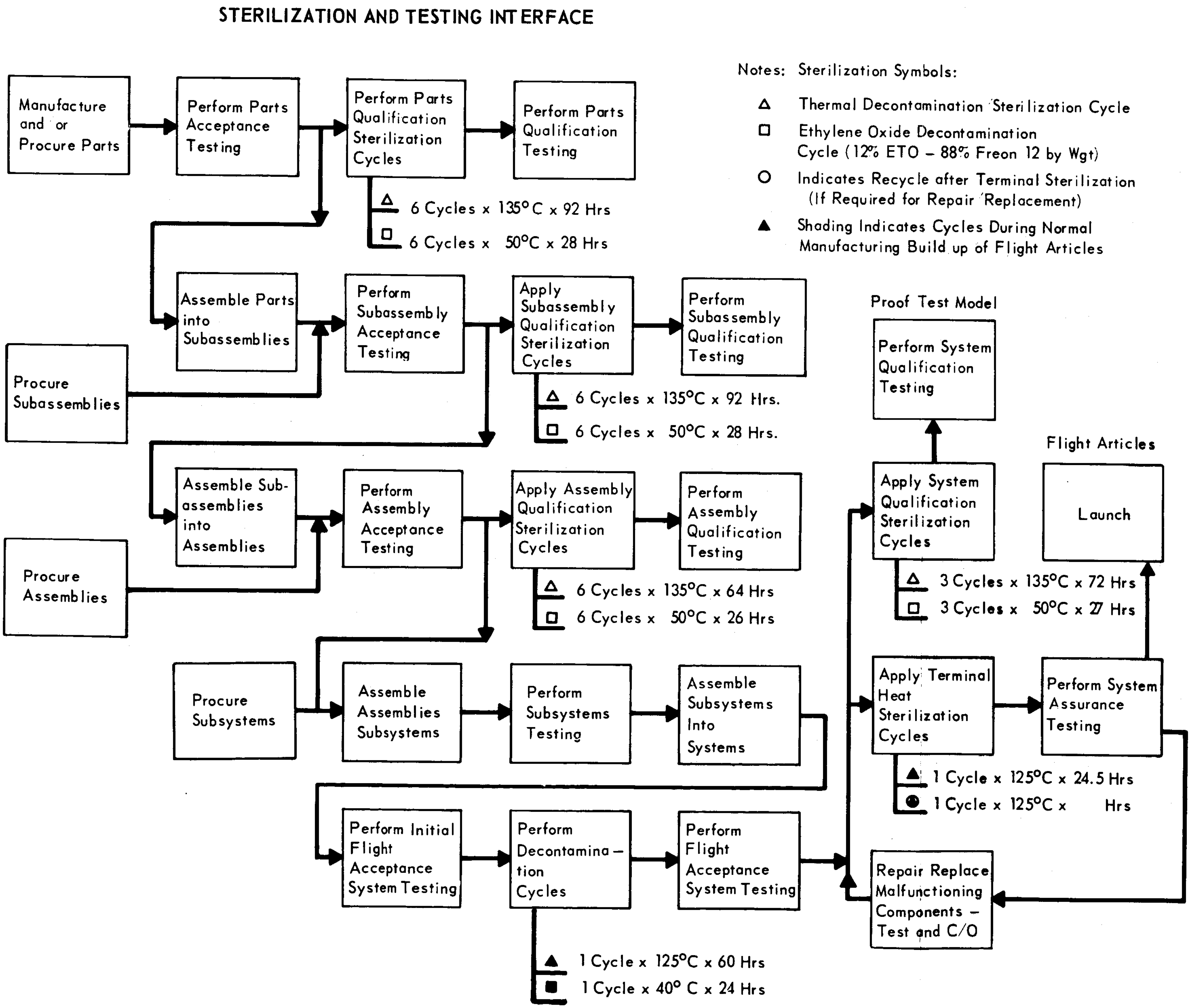

STERILIZATION AND TESTING INTERFACE
Notes: Sterilization Symbols:
Thermal Decontamination Sterilization Cycle
Ethylene Oxide Decontamination Cycle (12% ETO – 88% Freon 12 by Wgt)
Indicates Recycle after Terminal Sterilization (If Required for Repair Replacement)
Shading Indicates Cycles During Normal Manufacturing Build up of Flight Articles
Manufacture and or Procure Parts
Perform Parts Acceptance Testing
Perform Parts Qualification Sterilization Cycles
Perform Parts Qualification Testing
6 Cycles x 135°C x 92 Hrs
6 Cycles x 50°C x 28 Hrs
Assemble Parts into Subassemblies
Perform Subassembly Acceptance Testing
Apply Subassembly Qualification Sterilization Cycles
Perform Subassembly Qualification Testing
Procure Subassemblies
6 Cycles x 135°C x 92 Hrs.
6 Cycles x 50°C x 28 Hrs.
Proof Test Model
Perform System Qualification Testing
Assemble Sub-assemblies into Assemblies
Perform Assembly Acceptance Testing
Apply Assembly Qualification Sterilization Cycles
Perform Assembly Qualification Testing
Procure Assemblies
6 Cycles x 135°C x 64 Hrs
6 Cycles x 50°C x 26 Hrs
Flight Articles
Apply System Qualification Sterilization Cycles
Launch
3 Cycles x 135°C x 72 Hrs
3 Cycles x 50°C x 27 Hrs
Procure Subsystems
Assemble Assemblies Subsystems
Perform Subsystems Testing
Assemble Subsystems Into Systems
Apply Terminal Heat Sterilization Cycles
Perform System Assurance Testing
1 Cycle x 125°C x 24.5 Hrs
1 Cycle x 125°C x Hrs
Perform Initial Flight Acceptance System Testing
Perform Decontamination Cycles
Perform Flight Acceptance System Testing
Repair Replace Malfunctioning Components – Test and C/O
1 Cycle x 125°C x 60 Hrs
1 Cycle x 40° C x 24 Hrs

Figure 6–2

SECTION 7

RELIABILITY

The preferred design of the VOYAGER Capsule meets realistic reliability goals and requirements. In designing for reliability, we have emphasized three complementary approaches:

a. Conservative and simple design concepts, using proven techniques and processes.

b. Failure mode analysis to identify critical and potentially catastrophic effects.

c. Experienced engineering judgment, supplemented by a System Effectiveness Analysis, to optimize the choice for added equipment redundancies.

The estimated reliability of performing all VOYAGER Capsule mission functions by our preferred design is as follows:

	Equipment Reliability P_s	Experiments & Science Reliability P_s		Total P_s
Capsule Bus System	.830			.830
Entry Science Package	.957	.941		.901
Surface Laboratory	.891	.871		.776
	.708	.819	Total Capsule Reliability	.580

These estimates must be interpreted in their proper context. First, the Flight Capsule reliability of .580 refers to complete mission success, including successful operation of <u>all</u> experiments. Second, the probability of achieving less than <u>complete</u> mission success - for example, not all experiments are 100% successful - is considerably higher. Third, our experience indicates that predicted reliability values are generally much lower than those demonstrated; for example, the calculated reliability of Mariner IV was .111 per Planning Research Corporation Report PRC R-362 "Reliability Assessment of the 1964 Mariner Mars Spacecraft" dated 22 July 1963.

<u>Failure Mode Analysis</u> - The Failure Mode Effect and Criticality Analysis (FMECA) analyzes the probability of successfully achieving the major events required for mission success and evaluates their criticality. Alternate paths are identified that will circumvent potential failure of primary modes. This technique has pointed

out many failure conditions that are potentially critical and we have provided in our basic design alternate path functional redundancy for 63 mission events of the Capsule Bus System, 12 mission events of the Entry Science Package, and 59 mission events of the Surface Laboratory. Typical examples of alternate path functional redundancy for all three systems are shown in Figure 7-1.

System Effectiveness Analysis - The incorporation of equipment redundancies, as opposed to alternate path functional redundancies, has been guided by a System Effectiveness Analysis which identifies the most desirable order of redundancy incorporation. The analysis examines the change in the reliability of achieving specific mission objectives resulting from each redundancy and compares it to the incremental weight added. As part of this analysis, values were assigned to the three primary VOYAGER Capsule objectives.

a. Achievement of Flight Capsule Landing .4
b. Performance of Entry Science Experiments .35
c. Performance of Landed Science Experiments .25

These values are consistent with the priority established by the VOYAGER Mission General Specifications.

Figure 7-2 summarizes the results from the analysis on a capsule level. As noted, the "non-redundant" baseline design has a reliability of .37. This includes an estimate for the reliability of the science instruments in both the Entry Science Package and the Surface Laboratory. (The curve is discontinuous because the reliability trends differ for the Capsule Bus, the Entry Science Package and the Surface Laboratory.) A significant gain in reliability can be achieved at a moderate increase in Flight Capsule weight through selected redundancies as identified in this analysis. For example, an additional 100 pounds increases the reliability of this design to 0.67.

7.1 USE OF REDUNDANCY - Using the techniques discussed above, we have included in our preferred design 73 pounds of equipment redundancy, either functional, multichannel, or block. The resultant Flight Capsule weight is 4,776 pounds.

The choice of type of redundancy and its assignment to either the Capsule Bus, the Entry Science Package, or the Surface Laboratory was made by engineering judgment, guided by the results of the System Effectiveness Analysis. The approach is indicated in Figure 7-3. Curve A-B represents the theoretical optimum placement of redundancy and is the same as Figure 7-2. Curve A-C-D is the actual procedure which was employed. The redundancies represented by the part A-C were incorporated by engineering judgment to eliminate potentially severe single failure modes.

CAPSULE BUS – TYPICAL ALTERNATE PATH FUNCTIONAL REDUNDANCY

EVENT	PRIMARY MODE	FUNCTIONAL REDUNDANCY
1. Release and separate forward section of canister.	Canister Programmer	Capsule Bus Sequencer and Timer
2. Initiate Capsule Bus guidance and control computer routine.	Capsule Bus Sequencer and Timer	Capsule Bus Radar Altimeter
3. Initiate descent TV camera sequencing.	Capsule Bus Sequencer and Timer	.05g Sensor or Capsule Bus Radar Altimeter
4. Turn on Capsule Bus landing radar.	Capsule Bus Radar Altimeter	.05g Sensor or Capsule Bus Integrated Acceleration Sensing Routine
5. Terminate Capsule Bus terminal propulsion motor burn and capsule attitude control electronics.	Capsule Bus Landing Radar	Capsule Bus Sequencer and Timer or Impact Sensor or Surface Laboratory Impact Sensor

ENTRY SCIENCE PACKAGE – TYPICAL ALTERNATE PATH FUNCTIONAL REDUNDANCY

EVENT	PRIMARY MODE	FUNCTIONAL REDUNDANCY
1. Turn on Entry Science Package Telemetry Subsystem	Capsule Bus Sequencer and Timer	.05g Sensor or Capsule Bus Radar Altimeter
2. Sense Mach 5 and initiate experiments	Entry Science Package Sensor	Capsule Bus Radar Altimeter

SURFACE LABORATORY – TYPICAL ALTERNATE PATH FUNCTIONAL REDUNDANCY

EVENT	PRIMARY MODE	FUNCTIONAL REDUNDANCY
1. Switch SLS Sequencer and Timer to landed mode.	SLS Impact Sensors	Capsule Bus Impact Sensors or Capsule Bus Sequencer and Timer
2. Turn on SLS low rate S-Band transmitter.	SLS Sequencer and Timer	Capsule Bus Sequencer and Timer or Mission Operations System
3. Start surface sample collection.	Science Data System	Mission Operations System
4. Start growth experiment	Science Data System	Mission Operations System

Figure 7–1

ALLOCATION OF REDUNDANCY BY SYSTEM EFFECTIVENESS

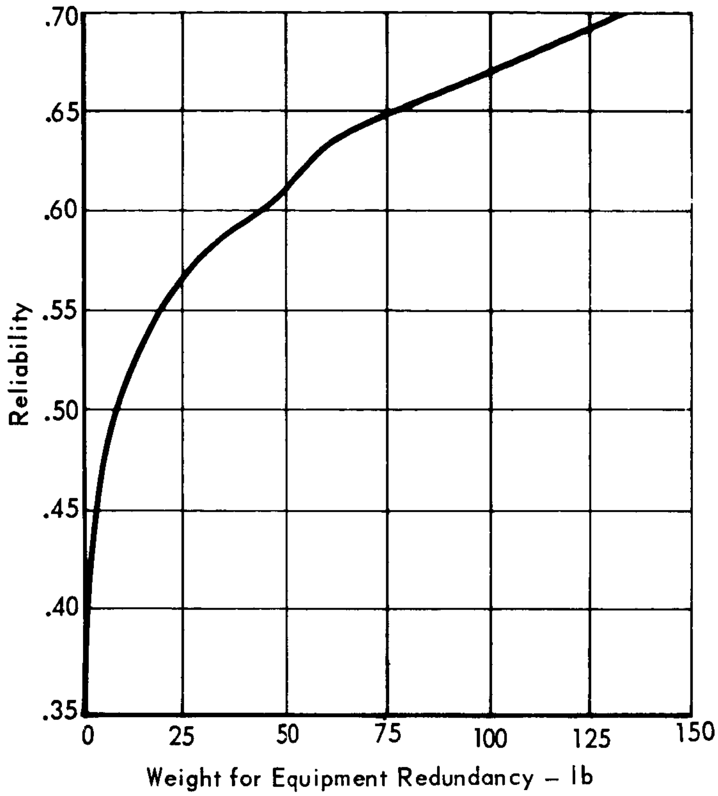


Figure 7-2

ALLOCATION OF REDUNDANCY FOR PREFERRED DESIGN

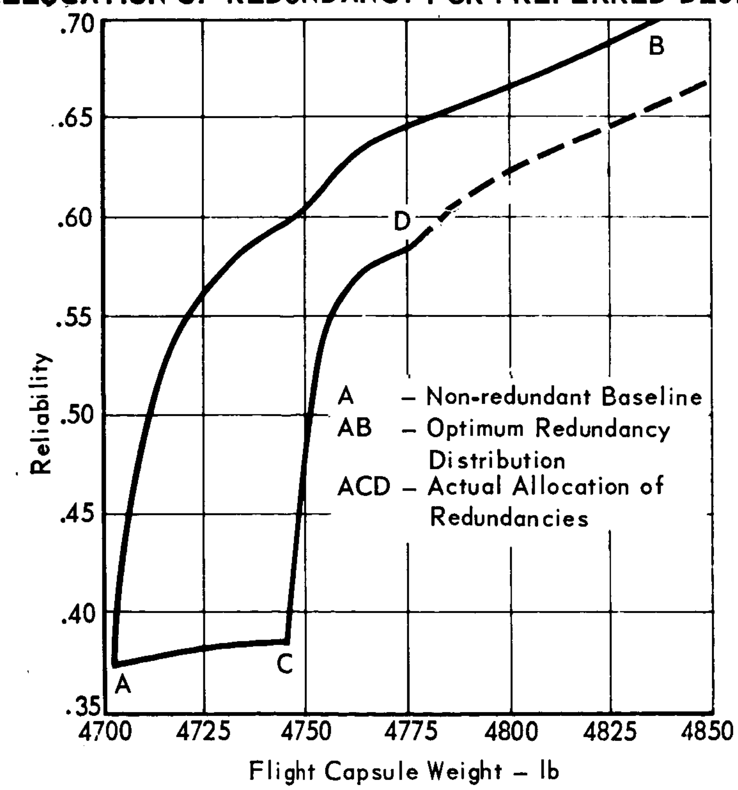


Figure 7-3

The remaining portion, C-D, represents redundancies incorporated in accordance with the System Effectiveness Analysis. The added 73 pounds increases the Flight Capsule reliability to 0.58. If we would have followed solely the analytical technique, without the application of engineering judgment, a predicted reliability as high as .648 could have been achieved, but in some cases, redundancies found to be important, based on our experience with Mercury, Gemini, and ASSET, would have been omitted.

The equipment redundancies that are part of our preferred design are listed in Figure 7-4 for the Capsule Bus, in Figure 7-5 for the Entry Science Package, and in Figure 7-6 for the Surface Laboratory.

7.2 POTENTIAL RELIABILITY IMPROVEMENT - The estimated weight of our preferred Flight Capsule design is 4776 pounds, and thus an additional 224 pounds is available as a weight margin which can be employed either to improve system reliability, or as weight contingency, or a combination of the two. Referring to Figure 7-3, it can be seen that further significant gains in Flight Capsule reliability can be achieved for a partial allocation of this weight margin to equipment redundancy. If about 75 pounds were applied in this manner, for example, the total system reliability could be increased to about 0.67.

REDUNDANCIES INCORPORATED IN CAPSULE BUS SYSTEM

SUBSYSTEM	REDUNDANCIES ADDED	TYPE	Δ WEIGHT (lb.)	REASON FOR INCORPORATION
1. Electrical Power	Active redundant dc-dc converter regulator	Multichannel	4.0	Provide redundancy of a critical function.
2. Telecommunications	CBS cruise encoder	Block	.15	Improvement of cruise data monitoring reliability – effective weight vs reliability addition.
3. Telecommunications	Series active redundant CBS cruise commutator, data switches and switch drivers	Multichannel	.45	Prevent short circuit failure modes which would cause loss of cruise engineering data.
4. Staging	Reefing cutters (2) for parachute reefing line.	Multichannel	.32	One of three reefing cutters required and enhances even opening.
5. Telecommunications	Series active redundant CBS cruise monitor control data switches and switch drivers.	Multichannel	.3	Improvement of flight capsule data monitoring reliability.
6. Guidance and Control	Fourth landing radar velocity sensor channel.	Multichannel	5.0	Three of four channels required for proper terminal descent control.
7. Telecommunications	Series active redundant adapter cruise commutator, data switches and switch drivers and standby redundant cruise encoder.	Multichannel and block	.6	Improvement of adapter cruise data monitoring reliability and prevent short circuit failure mode which would cause loss of adapter engineering data.
8. Guidance and Control	Active redundant receivers and trackers in radar altimeter	Multichannel	4.3	Provide altitude measurement backup for entry science correlation and optimum decelerator deployment.
9. Telecommunications	Multichannel cooperative redundant CBS radio link (interleave low rate CBS data on ESP radio link)	Multichannel	.25	Provide redundant method for retrieval of Capsule Bus data
10. Staging	Cartridge to explosive bolt assembly – capsule bus/adapter separation	Multichannel	2.96	Provide redundancy of a very critical event.
11. Staging	Cartridge to explosive bolt assembly – de-orbit motor release.	Multichannel	1.48	Provide redundancy of a very critical event.
12. Staging	Cartridge to explosive bolt assembly – Aeroshell release.	Multichannel	1.48	Provide redundancy of a very critical event.
13. Staging	Initiator in parachute catapult	Multichannel	.25	Provide redundancy of a very critical event.
14. Electrical Power	Relays and voltage sensors	Multichannel and block	1.25	Provide adequate power as and where needed without additional battery weight.
15. Staging	Cartridge to explosive bolt assembly – parachute release	Multichannel	1.48	Provide redundancy to assure release of parachute which if not released could prevent stabilization of capsule Lander after landing or could cover SLS.
16. Guidance and Control	Transmitter tube in radar altimeter	Block	1.44	Provide altitude measurement backup for entry science correlation and optimum decelerator deployment.
17. Staging	Shielded mild detonating cord assembly – forward canister release.	Multichannel	15.0	Provide redundancy of a very critical event.
18. Electrical Power	Third squib battery	Multichannel	8.5	Two of three are required during entry, therefore provides backup for critical events.
	Total added weight		49.2 lb.	

Figure 7-4

REDUNDANCIES INCORPORATED IN ENTRY SCIENCE PACKAGE

SUBSYSTEM	REDUNDANCIES ADDED	TYPE	Δ WEIGHT (lb.)	REASON FOR INCORPORATION
1. Telecommunications	ESP Cruise Encoder	Block	.15	Improvement of cruise data monitoring reliability – effective weight to reliability addition.
2. Telecommunications	Multichannel cooperative redundant ESP radio link (interleave low rate ESP data on CBS radio link).	Multichannel	.25	Provides redundant method for retrieval of low rate entry science and engineering data.
3. Telecommunications	Series active redundant ESP cruise commutator, data switches and switch drivers.	Multichannel	.45	Prevent short circuit failure mode which would cause loss of engineering data.
4. Electrical Power	ESP backup need voltage sensor and relay	Block	1.25	Provide adequate power as and where needed without additional battery weight.
	Total Added Weight		2.1 lb.	

Figure 7–5

REDUNDANCIES INCORPORATED IN SURFACE LABORATORY SYSTEM

SUBSYSTEM	REDUNDANCIES ADDED	TYPE	Δ WEIGHT (lb)	REASON FOR INCORPORATION
1. Telecommunications	SLS Cruise Encoder	Block	.15	Improvement of cruise data monitoring reliability – effective weight to reliability addition.
2. Telecommunications	Series Active SLS Cruise Commutator, Data Switches and Switch Drivers	Multichannel	.45	Prevent short circuit, failure mode which would cause loss of cruise engineering data.
3. Telecommunications	Functional Redundant SLS Low Rate Radio Link	Functional	6.3	Assure transmission of some surface and diagnostic data after landing.
4. Telecommunications	Functional Redundant SLS High Gain Antenna Pointing and Steering (Monopulse Tracking)	Functional	9.5	Provide backup of gyro-compassing mode of operation.
5. Staging	Cartridges in Pyrotechnic Devices (9) Surface Laboratory Experiment Deploy and Release	Multichannel	3.25	Increase reliability of obtaining experimental data with a small increase in weight.
6. Electrical Power	Relays	Multichannel	1.5	Increase reliability of flight capsule electrical power system.
7. Telecommunications	Sun Sensor	Functional	.4	Increase reliability of high gain antenna tracking capability.
	Total Added Weight		21.6 lb.	

Figure 7–6

SECTION 8

OPERATIONAL SUPPORT EQUIPMENT

The purpose of the Flight Capsule operational support equipment (OSE) is to provide - on a systems basis - the data to assess the functional adequacy and flight readiness of the capsule system and its subsystems in order to maximize the probability of mission success and to assure launch on time. We have selected an OSE approach which meets all VOYAGER program objectives and is compatible with the capsule system requirements and constraints, including integration of the capsule with the Flight Spacecraft and other VOYAGER systems. The choices are based on systems oriented analyses, in order to achieve a balanced approach with due consideration for schedule and cost objectives.

8.1 KEY REQUIREMENTS - Our analysis has identified five dominant requirements for OSE design. Our solutions to these requirements are as follows:

a. The Inviolate Launch Window - solution:

- Use the speed, repeatability, and safety of computer controlled checkout, but retain the man-in-the-loop for decision making and contingency action.

b. No Capsule Access After Canister Installation - solution:

- Integrate the system level test requirements with flight telemetry and in-flight checkout systems, and add an OSE umbilical, carrying selected critical parameters.
- Provide a System Test Complex (STC) that is capable of either automatic or manual system level testing, with minimum dependence on Subsystem Test Sets (SSTS). This approach provides maximum STC mobility and schedule flexibility, reduces OSE quantity and cost, and requires less space in the integrated control room.
- Selectively automate the SSTS to provide maximum test quality and repeatability of testing. This reduces the probability of a malfunction after canister installation and increases the validity of the subsystem test history for subsequent trend analysis and diagnosis.

c. Data Link at Launch Site (KSC) - solution:

- Provide a Ground Data Transmission System (GTDS) which uses low-error Bose-Chaudhuri coding techniques to transmit the multiple format composite test and command data over a single A2A and existing telephone lines at KSC. This results in minimum interference with other VOYAGER systems.

- o Convert the MFSK RF data for frequency domain multiplexing and transmission of these data over a single A2A line to the Telemetry Command Processor (TCP) computer in the STC for processing.
- o Use the spacecraft fly-away umbilical at the launch pad to carry the RF test data via coax cable from the CBS/SLS/ESP to the ground data transmission terminal at the vehicle.

d. <u>Deep Space Net Capability to Process SLS Telemetry</u> - solution:

- o Because of the potential saturation of existing computers at the DSIF, augment the computing capability by adding an SDS 930 or Sigma 5 computer.
- o As an alternative, use a special purpose computer to pre-process the SLS MFSK and convolution coded data to a level compatible with the computational capability of the existing SDS 920 TCP computers.

e. <u>Integration of SLS and ESP Hardware and Software into the CBS Test Complexes</u> - solution:

- o Establish during Phase B a foundation for Phase C allocation of the CBS/SLS/ESP contractor's hardware and software responsibilities and associated interface definition, by:
  - (1) Identifying each system's requirements separately.
  - (2) Identifying candidate equipment for integration of functions and time sharing.
  - (3) Preparing a preliminary plan for integrated software management which will be expanded during Phase C to provide central control of CBS/SLS/ESP software interfaces, programming, and functional integration.

8.2 <u>PREFERRED OSE APPROACH</u> - The preferred approach for the OSE of the Flight Capsule system is summarized in Figure 8-1. Unless noted, the design characteristics for each of the separate systems - CBS, ESP, and SLS - are identical. The similar nature of the CBS and SLS flight systems results in much the same OSE design approach and equipment requirements for both. The Entry Science Package OSE differs, however, because the management of the ESP will be by either the CBS or SLS contractors, who can include many of the ESP requirements into his System Test Complex.

OSE DESIGN CHARACTERISTICS SUMMARY

SUBSYSTEM TEST EQUIPMENT (SSTE)	• Direct analog hookup to flight subsystems. • Digital displays + hard copy print out. • Common design usable at all test sites. • Selected subsystem test sets automated. • Manual backup capability. • OSE self check. • Automatic alarm monitoring of critical parameters. • Test mode and data time tagged and recorded for data bank.
SYSTEM TEST COMPLEX (STC)	• Central computer used for automatic test sequence control, data monitoring and evaluation. • CRT display + keyboard + hard copy print out in engineering units. • Manual backup capability. • System test at KSC without subsystem test sets. • TCP computer used for TM data processing. • OSE self check. • Automatic alarm monitoring. • Monitors pad operations plus CB storage area.
LAUNCH COMPLEX EQUIPMENT (LCE)	• Launch monitor console in LCC for launch conditioning of CB, SLS, ESP. • Uses STC for remote monitor of flight TM • Direct hardlines for critical data. • Uses S/C flyaway umbilical + RF data link for launch pad data transmission. • Hardwired automatic alarm and safeing of critical functions. • Fault isolation to capsule or OSE. • Provides emergency power to SLS, CB, and OSE.
MISSION DEPENDENT EQUIPMENT (MDE)	• Preprocessing of SLS MFSK data for compatibility with TCP computers. • Uses software for CB and ESP decommutation • Special purpose hardware preprocesses CB and ESP telemetry for compatibility with TCP.
ASSY, HDNG, SHIPPING AND SERVICING (AHSE)	• Transporter capable of air, barge or helicopter usage. • Basic handling modules plus adapters for multifunction usage. • Servicing equipment mobile and self contained. • Provides emergency propellant dump at launch pad and ESF.
SOFTWARE	• Building block approach to software packaging and development. • Centralized management of CB, SLS, ESP test software.

Figure 8-1

Our checkout concept at KSC is shown in Figure 8-2. We consider this period of testing as most critical because of the limited launch window available, the complexity of the VOYAGER systems integration at KSC, and the requirement of not violating the sterility of the Flight Capsule during pre-launch checkout activities.

8.3 OSE EQUIPMENT - The major OSE equipment requirements are listed in Figure 8-3. Additional OSE components will be required beyond those included, but these are of lesser magnitude and complexity. The total quantity of OSE items is 87 for the CBS, 51 for the SLS, and 24 for the ESP.

8.4 IMPLEMENTATION - We have identified the major pacing items and significant OSE events on Figure 8-4. Critical lead items are:

a. The STC computers must be ordered in December 1968, during Phase C, to allow for an 11 month delivery lead time.

b. Detailed test software and the balance of STC equipment must be available by June 1970 in time for compatibility tests with the Integrated Systems Bench Test Unit (ISBTU).

c. Interface simulators and System Test Complex Equipment is needed from the SLS and Spacecraft contractors by November 1970 for CBS test complex installation and validation by June 1971.

CAPSULE BUS SYSTEM OSE – UTILIZATION

SUBSYSTEM TEST SETS
- Equipment Functional Checks
- Module & Subsystem Tests

SYSTEM TEST COMPLEX EQUIPMENT

STC
- Integrated Subsystems Tests
- Systems Assurance Test
- Simulated Mission Test
- Vibration Tests
- Environmental Tests
- CB Systems Assurance Tests
- PV Systems Assurance Tests
- PV Simulated Mission
- PV Simulated Mission (J Fact)
- Post Sterilization CB Assurance
- PV System Confidence
- Final PV System Assurance
- PV System Verification
- J Fact
- Countdown Demonstration
- Countdown

LAUNCH COMPLEX EQUIPMENT

LCE
- Ground Power
- UHF and S-Band Group
- Test Stimuli

MISSION DEPENDENT EQUIPMENT

MDE
- DSIF Compatibility
- Countdown
- Cruise Monitor
- Inflight Checkout
- Descent Monitor
- Landed Operations

ASSEMBLY HANDLING & SERVICING EQUIPMENT

AHSE
- Handling/Transportation
- Servicing
- Pyro, De-orbit Motor Installation

SPACECRAFT MOUNTED EQUIPMENT

SCME
- CB & ESP System Tests

FLIGHT CAPSULE CHECKOUT CONCEPT

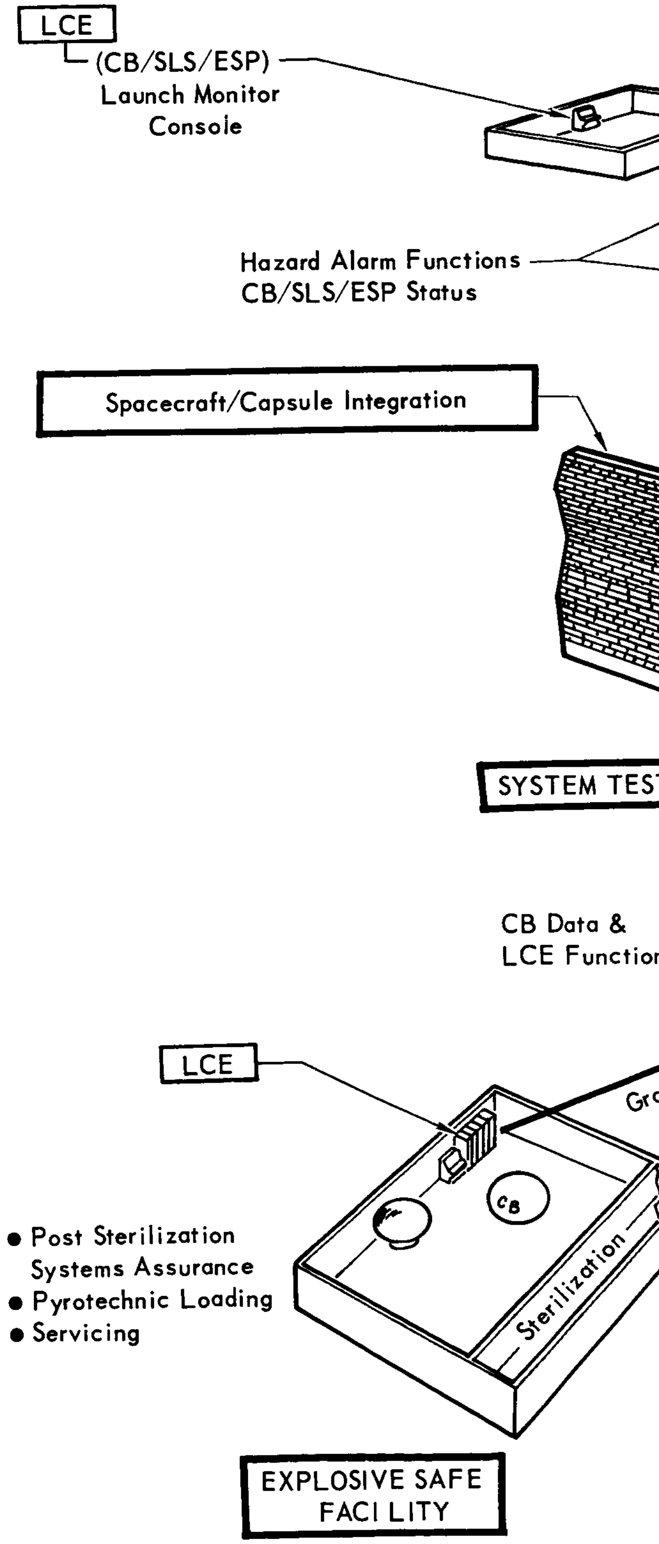
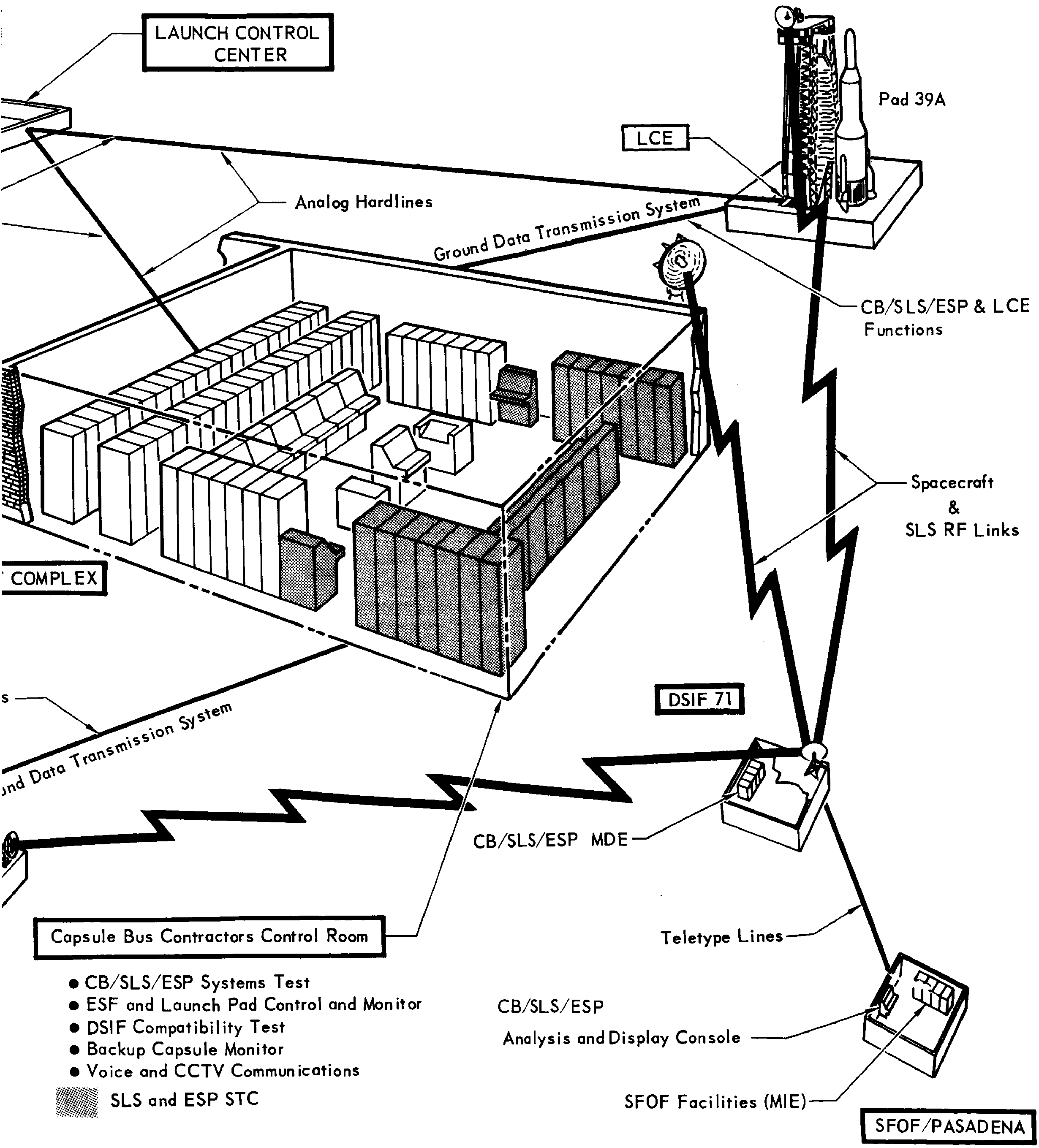


Figure 8-2

AT KSC



LAUNCH CONTROL CENTER
Pad 39A
LCE
Analog Hardlines
Ground Data Transmission System
CB/SLS/ESP & LCE Functions
Spacecraft & SLS RF Links
COMPLEX
DSIF 71
s
und Data Transmission System
CB/SLS/ESP MDE
Teletype Lines
Capsule Bus Contractors Control Room
● CB/SLS/ESP Systems Test
● ESF and Launch Pad Control and Monitor
● DSIF Compatibility Test
● Backup Capsule Monitor
● Voice and CCTV Communications
SLS and ESP STC
CB/SLS/ESP
Analysis and Display Console
SFOF Facilities (MIE)
SFOF/PASADENA

5-5-2

MAJOR OSE EQUIPMENT REQUIREMENTS

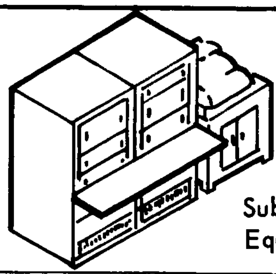
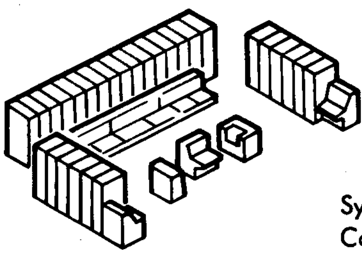
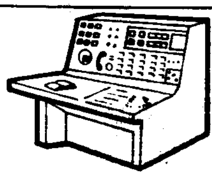
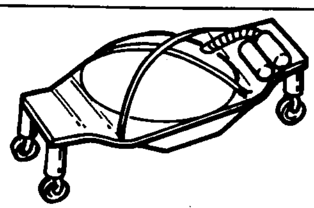
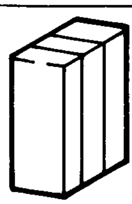
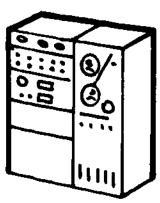
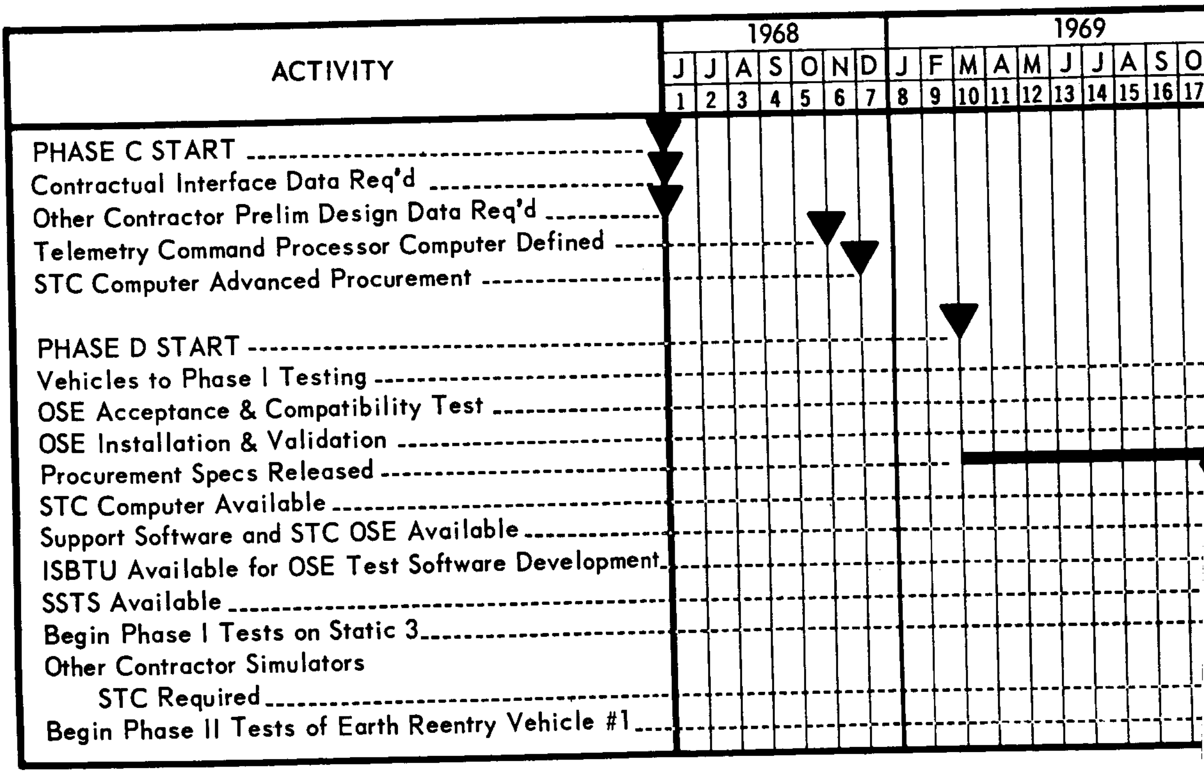
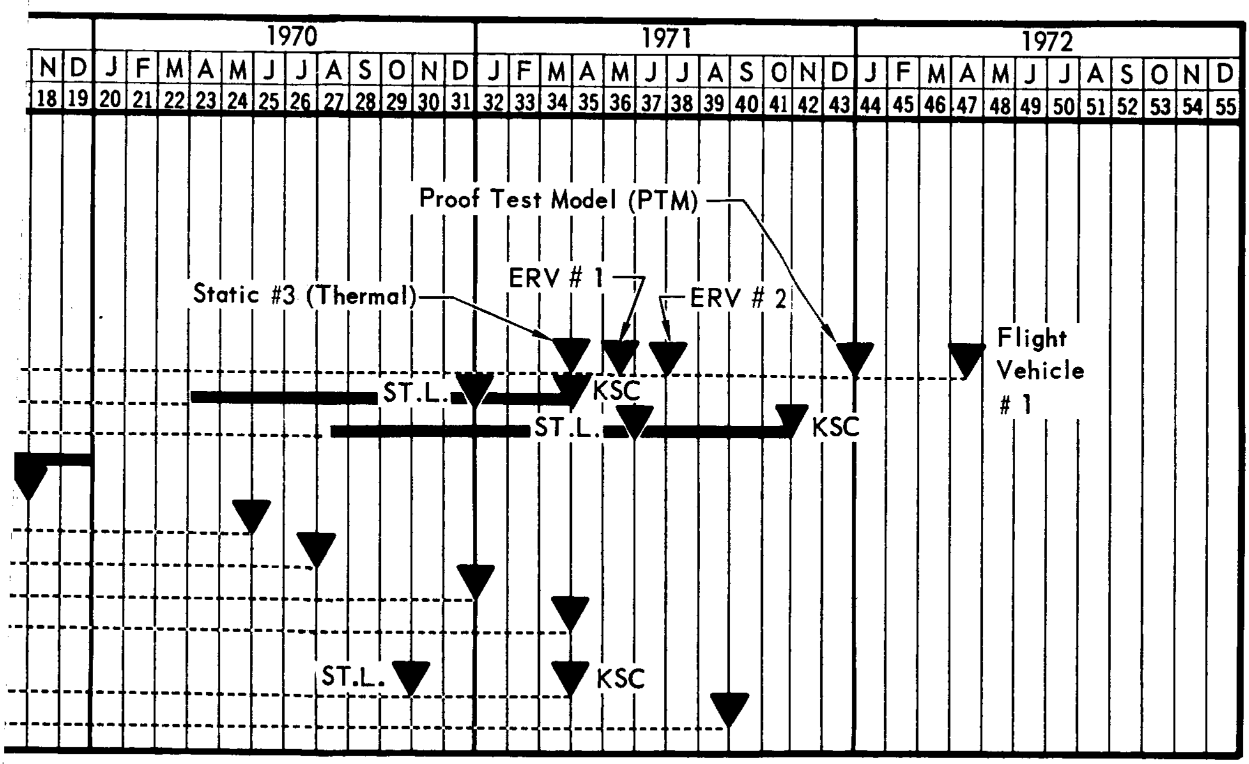
	OSE CLASS	OSE		APPLICATION CB	SLS	ESP
 Subsystem Test Equipment	SSTE	● Science	Test Set		●	●
		● Power	Test Set	●	●	●
		● Sequencer	Test Set	●	●	
		● TCM	Test Set	●	●	●
		● G & C	Test Set	●		
		● Propulsion	Test Set	●		
		● Canister & Adapter	Test Set	●		
		● Radar	Test Set	●		
 System Test Complex	STC	● Test Director's Console		●	●	
		● CB/SLS/ESP Subsystem Console		●	●	●
		● Timing, Intercom and CCTV		●	●	●
		● TCM Equipment		●	●	●
		● TCP Computer Equipment		●	●	●
		● CDS Computer Equipment		●	●	
		● Ground Data Transmission System		●	●	●
		● Computer Software		●	●	●
 Launch Complex Equipment	LCE	● Launch Monitor Console		●	●	
		● Ground Power & Dist. Console		●	●	●
		● Remote Stimuli Equip.		●	●	●
		● UHF or S-Band RF Groups		●	●	●
 Assembly, Handling, Shipping Equipment	AHSE	● Flight Capsule Transporter		●		
		● C.B. Handling Dolly		●		
		● C.B. Handling Fixture		●		
		● Fluid & Gas Servicing Units		●		
		● Capsule/Canister Assy & C/O Stand		●		
		● Propellant Purge & Disposal Units		●		
 Spacecraft Mounted Equipment	SCME	● SC Mounted TCM Subsystem Test Set		●		●
 Mission Dependent Equipment	MDE	● Data Demultiplexer		●		●
		● SFOF Display Console		●	●	●
		● Capsule Simulator		●	●	●
		● MFSK Detection Equipment			●	
Total Quantity – All OSE Items Identified				87	51	24

Figure 8-3

OSE IMPLEMENTATION SCHEDULE



8-7-1



1970
1971
1972
N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D
18 19 20 21 22 23 24 25 26 27 28 29 30 31 32 33 34 35 36 37 38 39 40 41 42 43 44 45 46 47 48 49 50 51 52 53 54 55
Proof Test Model (PTM)
ERV # 1
ERV # 2
Static #3 (Thermal)
Flight Vehicle # 1
ST.L.
KSC
ST.L.
KSC
ST.L.
KSC

Figure 8-4

SECTION 9

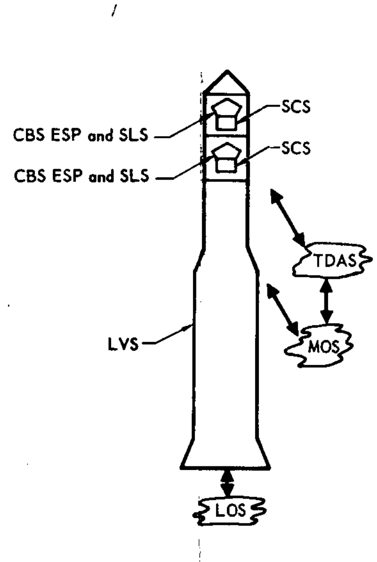
INTERFACES

The important interfaces of the Capsule Bus System, the Entry Science Package, and the Surface Laboratory System are delineated in Figure 9-1, including those with the other major systems of the VOYAGER Program. Some of the interfaces are functional - physical, electrical, mechanical, signals, etc. - and some concern software - documents, procedures, training, etc.

Of special significance is the interface complexity between the ESP and the other Capsule systems. We find that extensive ESP interfaces exist with the Capsule Bus. On the other hand, those with the Surface Laboratory are generally minor. In fact, the only major ESP interface with the Surface Laboratory is the back-up battery power which the SLS provides. Thus, from an interface standpoint alone, it would seem preferable to include the ESP as part of the Capsule Bus System, thus assuring an efficient integrated design which minimizes the interface problems.

CAPSULE BUS, ENTRY SCIENCE PACKAGE AND SURFACE LABORATORY TO OTHER SYSTEM INTERFACES

	CBS (Delivers ESP through Martian entry and SL to Martian surface.)	ESP (Conducts Martian entry measurements/experiments)	SLS (Martian Surface Science/Experiments)
ESP	● Physical mounting of ESP equipment Sensors. ● Signals associated with sequencing telemetry data, and commands (includes those routed through CB to SC). ● Thermal control of ESP by CB. ● OSE compatibility for pre-launch checkout.		● SL provides ESP with backup battery power.
SLS	● Installation provisions for SL in CB. ● Signals associated with sequencing, telemetry data and commands (includes those routed through CB to SC). ● Thermal control of SL by CB. ● OSE compatibility for pre-launch checkout.	● SL provides ESP with backup battery power.	
SCS (Delivers Flight Capsules into Martian Orbit)	● CB to SC structural field joint. ● Signals associated with sequencing, telemetry data, commands and inflight checkout. ● Power supplied to CB by SC. ● Spacecraft mounted support equipment – to provide inflight checkout, CB to SC RF relay link, SC sequence and timing commands and backup commands (includes SC power for this equipment). ● OSE logic and power levels, data formats, power regulation, source and load impedance, bit rate, hoisting and handling compatibilities. ● Maintenance of sterilization level – unsterilized SC mated with sterilized Flight Capsule.	● RF (data relay) and inflight checkout provisions.	● Power and inflight checkout provisions
LVS (Delivers Planetary Vehicles on Martian Trajectory)	● LV provides environmental control under nose fairing on inside the shroud after mate for planetary vehicles (temperature, humidity, cleanliness). ● Envelope constraints on CB.		
LOS (Launches Space Vehicle)	Provides physical and functional support to pre-launch and launch activities.	● Provides functional support to pre-launch activities.	● Provides physical and functional support to prelaunch and launch activities.
TDAS (Acquires data, tracks, and transmits commands)	● Telemetry (via SC) to all Voyager TDAS stations. ● MDE requirements for space, power, and signal interconnections. ● Flight capsule simulator requirements – same as for MDE. ● Tracking information for pre-landed mission operations.	● Telemetry and command signals via SC S-band downlink.	● Telemetry and command data streams (S-Band downlink and uplink). ● Tracking information for mission operations. ● MDE requirements for space, power and signal interconnections (includes provisions for Flight Capsule simulator).
MOS (Operations)	● Functional support for command data stream (MOS teams originate and verify commands that cross MOS/TDAS and TDAS/SCS interfaces through SC to the CB. ● Functional support for telemetry data stream (data command verification is fed back from CB through SC to MOS; starting in CB, it crosses SCS/TDAS and TDAS/MOS interfaces before reaching MOS teams). ● Data from CB system tests provided MOS. Computer data reduction and analysis programing. ● Training and procedural interfaces (includes MDE operation by MOS personnel and contingency plans). ● Post-launch decisions and operations.	● SFOF analysis of entry science data for mission operations. ● Computer programs for mission analysis.	● Functional support for command data stream (MOS teams originate and verify commands that cross MOS/TDAS and TDAS/SLS interfaces. ● Functional support for telemetry data stream (reverses the above interfaces). ● Computer programs for data reduction and analysis. ● Training and procedural interface (includes MDE operation by MOS personnel, and contingency plans). ● SFOF mission analysis.



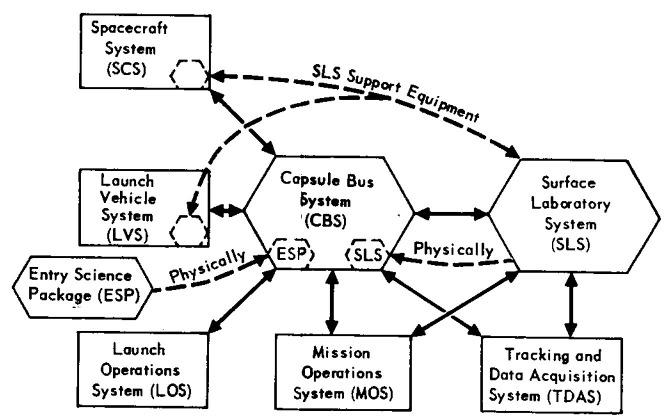


Figure 9–1

SECTION 10

IMPLEMENTATION

A key requirement for successful development of the VOYAGER Flight Capsule is a thoroughly integrated and dynamic plan for action. The events and activities of this plan outline the effort required to accomplish the development of the Capsule Bus, Surface Laboratory, and Entry Science Package.

Five major constraints are significant to this program, each imposing stringent requirements for detailed and meticulous implementation planning.

a. The Inflexible Launch Period - Precise schedules must be established and proper controls exercised. At the same time, these schedules must contain sufficient flexibility for contingencies.

b. Planetary Quarantine - The demands of ultra-clean assembly, microbiological monitoring, and sterilization will impose additional time and costs, and will increase the need for precise planning.

c. Uncertain Martian Environment - Initial parallel concept development may be mandatory for some critical subsystems, to accommodate later definitions of environmental constraints.

d. Experiment Integration - Science instruments and experiments must be closely coordinated with the vehicle development schedule via an experiment integration plan.

e. Interfaces - Coordination with other interfacing VOYAGER systems will require a constant information flow regarding hardware, software, and operations.

We have examined all the aspects involved in implementing the Flight Capsule systems. The key elements of our approach are:

a. Factory-To-Pad Delivery of Assembled Vehicles - Once the Flight Capsule has completed its factory flight acceptance tests, it is shipped intact to the launch site. The idea here is: Test thoroughly and once you connect it - don't disconnect.

b. Hardware Qualification - All hardware is qualified prior to delivery of vehicles to the launch site.

c. Life Testing - All nonmetallic materials are qualified for long life under simulated space and Mars environment for 43 weeks. All equipment which is not required to operate until just before Capsule separation is exposed to a four-week "operating life" test that simulates the nonoperating interplanetary cruise as well as the active periods.

d. Flight Capsule Engineering Model - Subsystem compatibility tests are performed on an engineering model of the total Flight Capsule (which includes all subsystems of the CBS, SLS, and ESP) early in the development program, for early identification of interface problems. Our schedule requires that engineering models of subsystems be available in April 1970 and that compatibility testing start in July 1970.

e. Flight Proof Tests - We find that an Earth reentry vehicle test program is very desirable in order to proof test the Canister and Aeroshell separation techniques and the operation of the terminal deceleration and guidance system of the lander at Earth altitudes corresponding to Mars environments (above 120,000 feet). These functions cannot be adequately simulated on the ground. However, because a Saturn IB - or equivalent - booster is required, the cost of such a flight test program is unduly high. Therefore, such testing would have to be justified from an overall VOYAGER Program view, and not just from the viewpoint of Capsule development.

f. Contingency Planning - Contingencies are incorporated in our Master Schedule, such as a 10% time contingency for in-house manufacture; a 13-week contingency period for vendor hardware predelivery acceptance tests and equipment functional checks; delay of the initiation of hardware fabrication until 50% of the qualification testing for each hardware item has been completed; and provision for a Capsule recycle capability at the launch site.

The implementation Master Schedule, based on a PERT analysis of time-phased events, indicates that, for a Phase D go-ahead of 1 March 1969, the only major subsystem that is sufficiently time-critical to require early go-ahead in Phase C is the Capsule Bus terminal propulsion subsystem. It requires initiation of development approximately 18 weeks before the start of Phase D. The next most critical items are the sequencer and timer, data storage, and telemetry for the Surface Laboratory, each with a negative slack of seven weeks with respect to the start of Phase D.

The development schedules for each of the three major systems - the Capsule Bus, the Entry Science Package, and the Surface Laboratory System - are discussed below.

10.1 CAPSULE BUS SYSTEM - The Master Schedule for the Capsule Bus is shown in Figure 10-1. Time-critical subsystems are:

Critical Subsystem	Approximate Weeks Criticality	Critical Item
Terminal Propulsion	-18	Component development and testing of rocket engines
Reaction Control	- 6	Component and development testing of rocket engines
Data Storage	- 5	Development and testing of memory core units
Telemetry	- 4	Programmer development and testing
Radio	- 3	Selection of transistors and development and testing of exciter/power amplifier

We don't consider negative slack time periods of greater than six weeks as being significant at this time, because of the estimation methods on which the PERT analysis is based. The table does indicate, however, that the development for the above subsystems must be started essentially at the time of Phase D go-ahead, except for the terminal propulsion subsystem, which must be started even earlier.

10.2 ENTRY SCIENCE PACKAGE - The Master Schedule for the Entry Science Package is shown in Figure 10-2. Critical subsystems are:

Critical Subsystem	Approximate Weeks Criticality	Critical Item
Telemetry	- 4	Programmer development and testing
Data Storage	- 3	Development and testing of memory core units
Radio	- 2	Development and testing of 400 MHz transmitter and bit synchronizer

As in the case of the Capsule Bus, the negative slack times indicate that these subsystems must be started essentially at the time of Phase D go-ahead.

10.3 SURFACE LABORATORY SYSTEM - The Master Schedule for the Surface Laboratory System is shown in Figure 10-3. Critical subsystems are:

VOYAGER CAPSULE BUS SYSTEM SUMMARY SCHEDULE

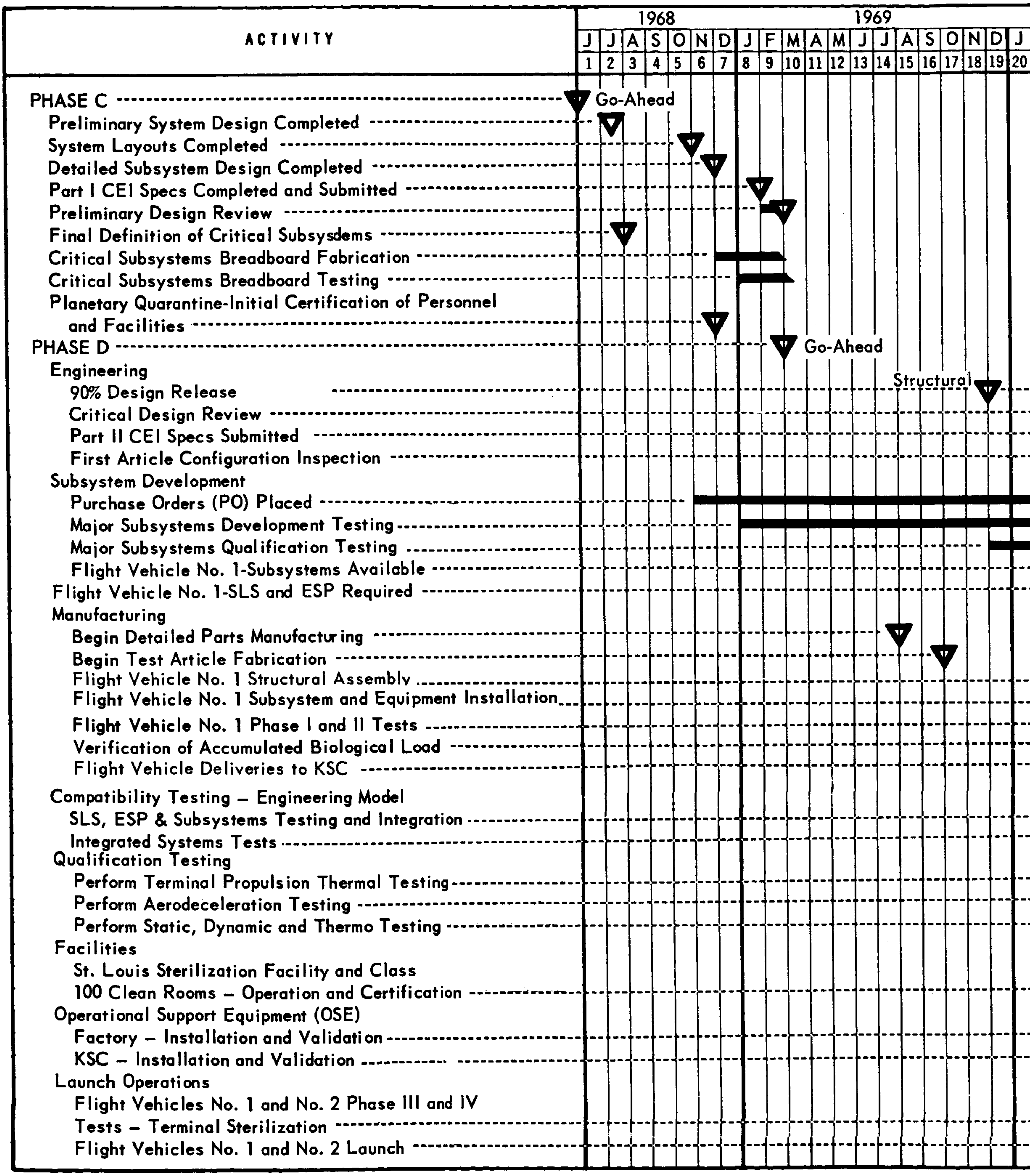
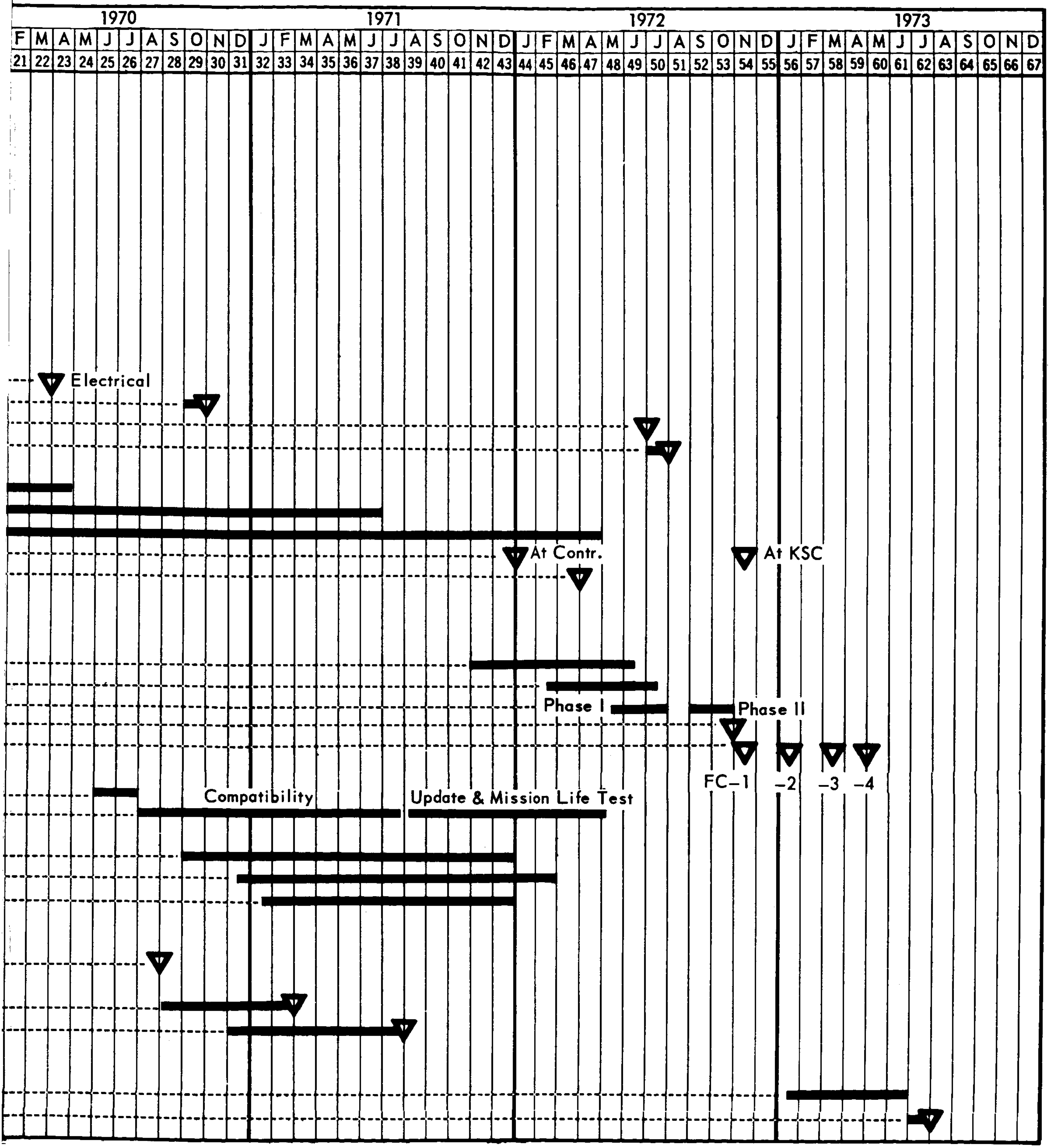


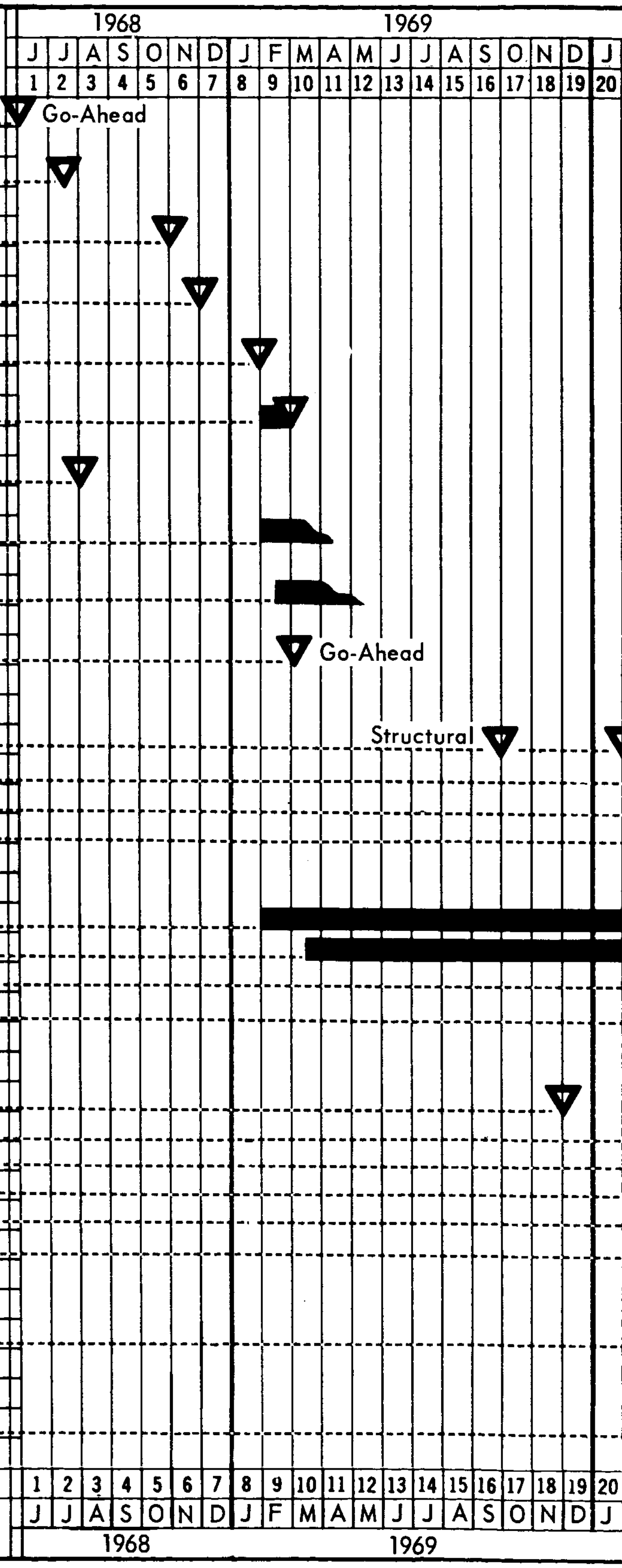
Figure 10-1



1970
1971
1972
1973
F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D
21 22 23 24 25 26 27 28 29 30 31 32 33 34 35 36 37 38 39 40 41 42 43 44 45 46 47 48 49 50 51 52 53 54 55 56 57 58 59 60 61 62 63 64 65 66 67
Electrical
At Contr.
At KSC
Phase I
Phase II
FC–1
–2
–3
–4
Compatibility
Update & Mission Life Test

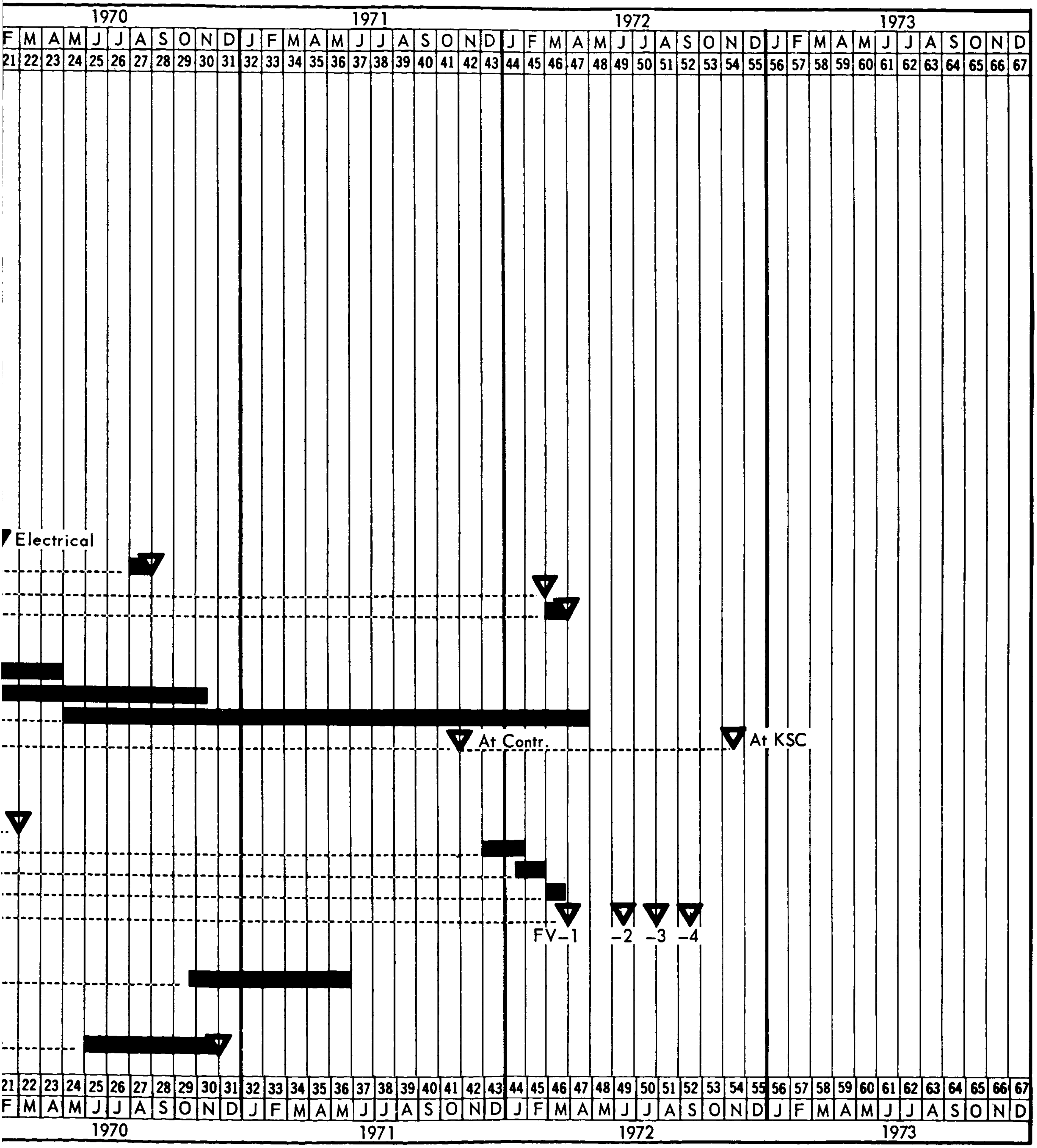
VOYAGER ENTRY SCIENCE PACKAGE – SUMMARY SCHEDULE

	ACTIVITY
1	PHASE C
2	
3	Preliminary System Design Completed
4	
5	System Layouts Completed
6	
7	Detailed Subsystem Design Complete
8	
9	Part I CEI Specs Completed and Submitted
10	
11	Preliminary Design Review
12	
13	Final Definition of Critical Subsystems
14	
15	Critical Subsystem Breadboard Fabrication
16	
17	Critical Subsystems Breadboard Testing
18	
19	PHASE D
20	
21	Engineering
22	90% Design Release
23	Critical Design Review
24	Part II CEI Specs Submitted
25	First Article Configuration Inspection
26	
27	Subsystem Development
28	Purchase Orders (PO) Placed
29	Subsystems Development Testing
30	Subsystems Qualification Testing
31	Flight Vehicle No. 1 Subsystems Available
32	
33	Manufacturing
34	Begin Detailed Parts Manufacture
35	Begin Test Article Fabrication
36	Flight Vehicle No. 1 Structural Assembly
37	Flight Vehicle No. 1 Subsystems & Equipment Instl
38	Flight Vehicle No. 1 Systems Tests
39	Flight Vehicle Deliveries to Integ Contr
40	
41	Qualification Testing
42	Perform Static, Dyanmic & Thermal Testing
43	
44	Operational Support Equipment (OSE)
45	Factory – Installation & Validation
46	

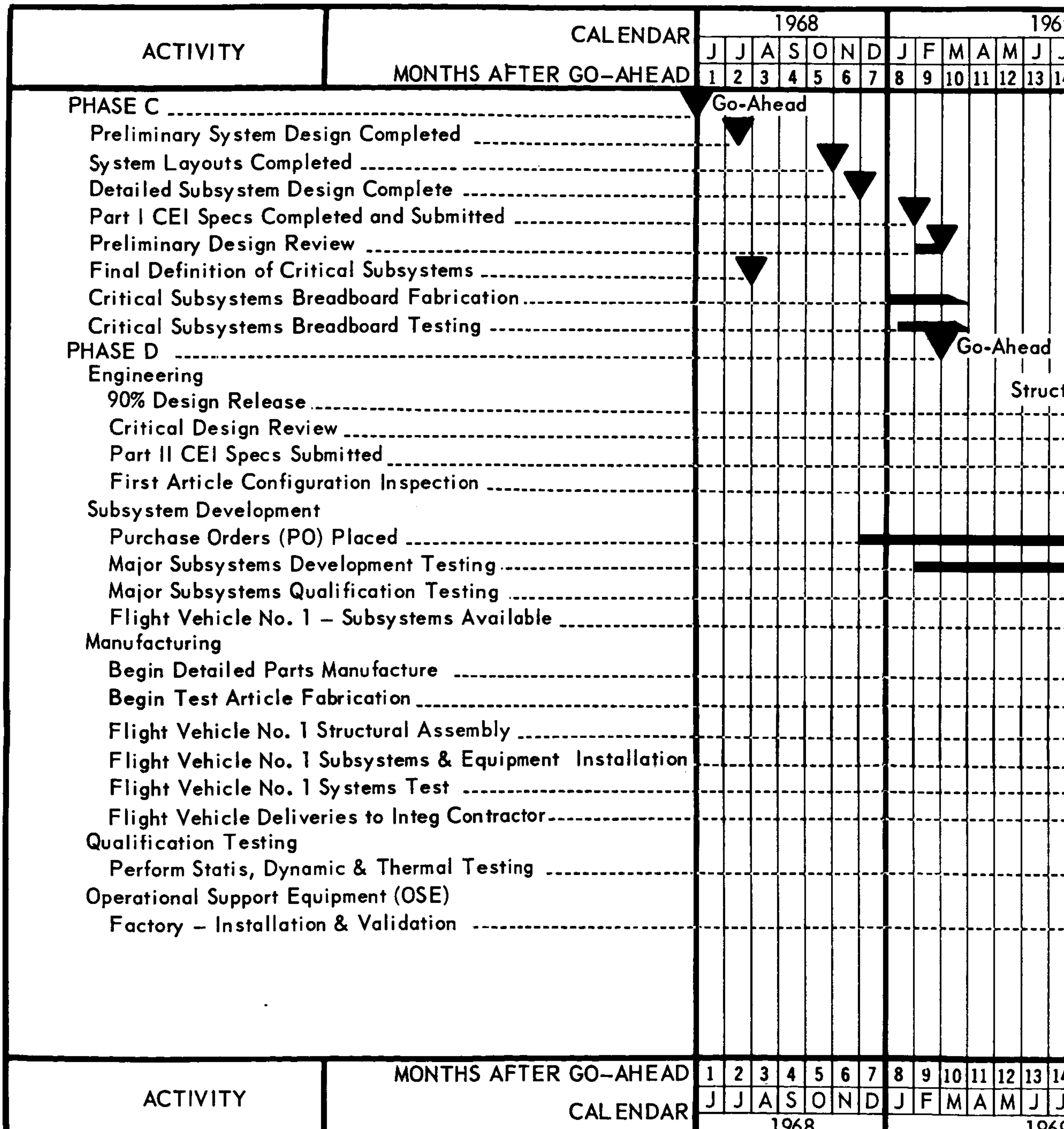


MONTHS AFTER GO-AHEAD
CALENDAR

Figure 10–2

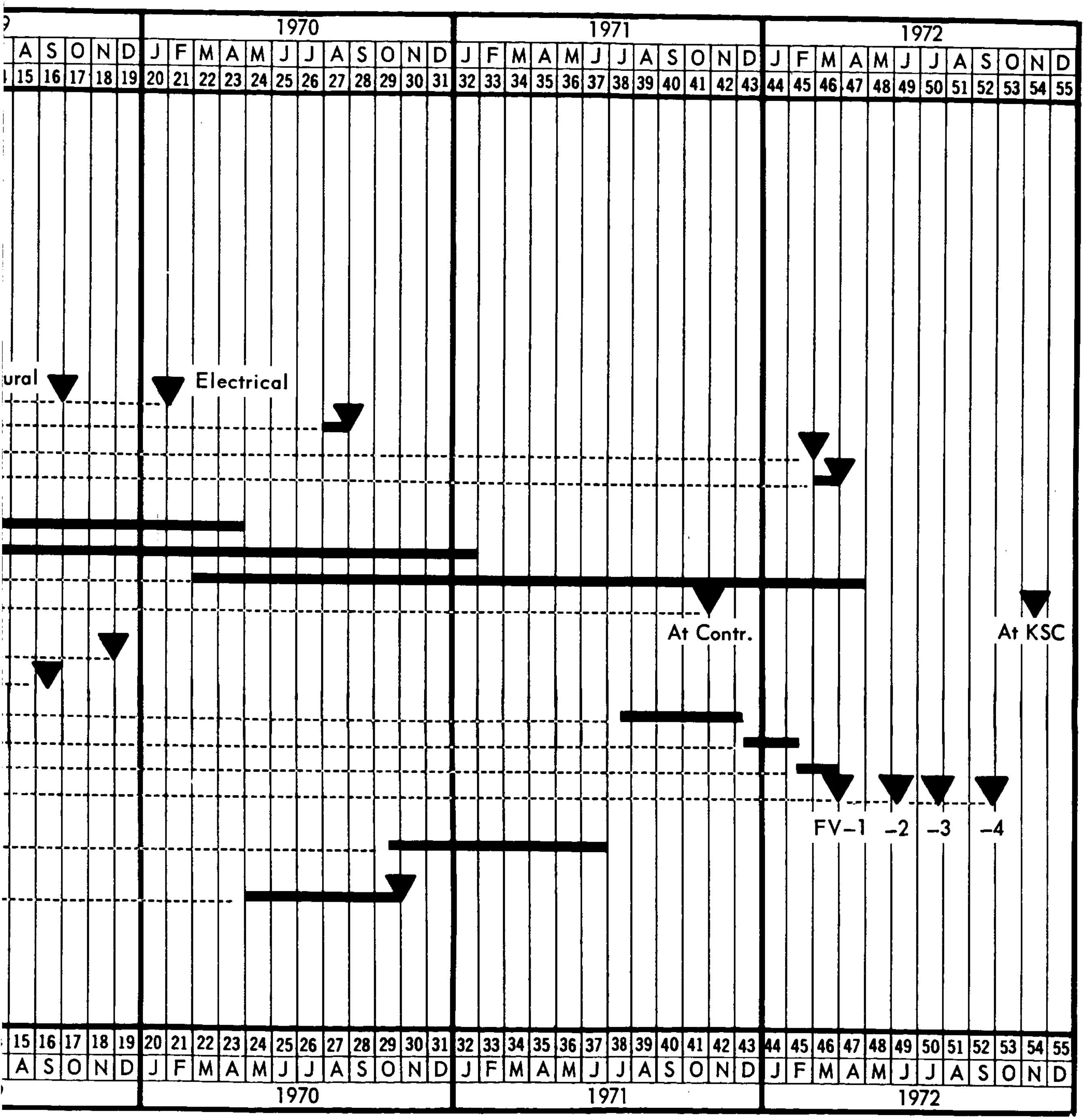

1970
1971
1972
1973
F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D
21 22 23 24 25 26 27 28 29 30 31 32 33 34 35 36 37 38 39 40 41 42 43 44 45 46 47 48 49 50 51 52 53 54 55 56 57 58 59 60 61 62 63 64 65 66 67
Electrical
At Contr.
At KSC
FV–1
–2
–3
–4

VOYAGER SURFACE LABORATORY SYSTEM
SUMMARY SCHEDULE



Figure

10-6-1


1970
1971
1972
ural
Electrical
At Contr.
At KSC
FV–1
–2
–3
–4

Critical Subsystem	Approximate Weeks Criticality	Critical Item
Sequencer & Timer	-7	Development and testing
Data Storage	-7	Development and testing of the tape recorder
Telemetry	-7	Development and testing of the programmer and experiment controller
Radio	-6	Development and testing of the low-rate radio
Thermal Control	-5	Development and testing of heat pipes
Command	-4	Command decoder development and testing

The sequencer and timer, data storage, and telemetry appear to be on the borderline of criticality, and development may have to be initiated during the latter stages of Phase C. As noted, the subsystem criticality for the Surface Laboratory is generally greater than for the Capsule Bus. This is due to the requirement for completion of the Surface Laboratory prior to its integration with the Capsule Bus.